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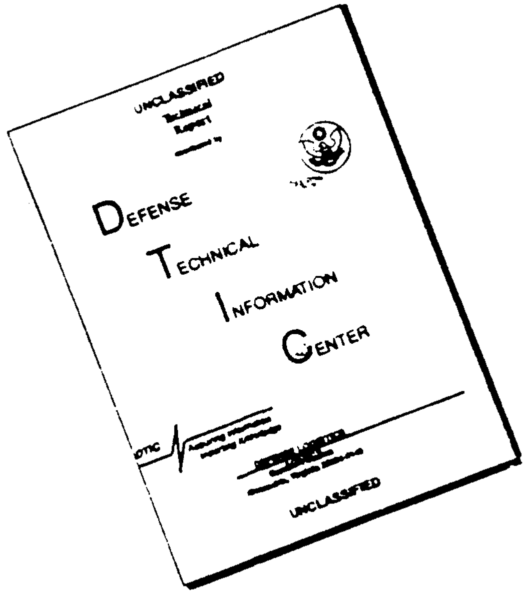

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Advanced Airframe Structural Materials

A Primer and Cost Estimating Methodology

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prepared for the
United States Air Force

RAND

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PREFACE

This report documents the results of a study addressing the implications of advanced materials on airframe structure cost. More specifically, an estimating methodology for overall airframe cost was developed that is sensitive to the material composition of the basic structure and suitable for use in a program's conceptual stage when little detailed design information is available. A primer describing and assessing new materials technology and related manufacturing processes is also presented.

This research was performed for the Deputy Assistant Secretary for Cost and Economics and was undertaken as part of a project entitled "Air Force Resource and Financial Issues" within the Resource Management and System Acquisition Program of Project AIR FORCE.

SUMMARY

This report identifies, describes, and quantifies the cost effects of structural materials that are likely to be incorporated into aircraft becoming operational in the 1990s (aluminum, aluminum-lithium, steel, titanium, graphite/epoxy, graphite/bismaleimide, and graphite/thermoplastic). The first half of this report is a primer for advanced aircraft structural materials emphasizing polymer matrix composites. The second half of the report contains both cost data and a cost estimating methodology sensitive to material mix. For each material type, separate cost factors are presented for two time frames, the late 1980s and the mid-1990s, and for the following cost elements: nonrecurring engineering, nonrecurring tooling, recurring engineering, recurring tooling, manufacturing labor, manufacturing material, and quality assurance. These factors are based on data obtained from Boeing Airplane Company, General Dynamics Corporation, Grumman Aerospace Corporation, Lockheed Aerospace Systems Corporation—California Division and Georgia Division, LTV Aerospace and Defense Aircraft Group, McDonnell Douglas Corporation, Northrop Aircraft Division, and Rockwell International Group.

The new materials and processes that are most likely to be utilized in aircraft structure will be aluminum-lithium, superplastically formed (SPF)/diffusion bonded (DB) titanium, and polymer matrix composites (for example graphite/epoxy).[1] The other categories of composite materials (ceramic matrix, metal matrix composites and carbon/carbon composites), powder metallurgy, and titanium-aluminum alloys are not likely to constitute substantial proportions of 1990s aircraft. These materials either (1) offer properties that are too specialized for widespread application to aircraft structure or (2) are behind polymer matrix composites or other metals in terms of development.

Two important categories of polymer matrix composites are thermosets and thermoplastics. All composites used up to now on production aircraft have been thermosets. However, continuous fiber reinforced thermoplastics have recently received considerable attention. The important distinction between the two is that thermoplastics, unlike thermosets, may be reheated and reshaped. This property offers enormous potential for processing ease. However, few engineering data exist regarding thermoplastics, creating uncertainty in both the design and fabrication processes.

Although composites clearly offer several performance advantages including reduced weight, reduced number of fasteners, increased corrosion resistance, reduced radar cross-section, and the potential for extended life, this increase in capability comes at a price. The cost data collected clearly show that composite materials are more expensive than aluminum on a cost-per-pound basis. In fact, in terms of *overall* recurring cost per pound,[2] composites are over two times more expensive than aluminum. A few of the reasons for the higher costs of composites are: (1) properties of these engineered materials frequently need to be verified, (2) tooling requires sophisticated design and durability because of the severe thermal cycling experienced in the autoclave, (3) current hand-layup techniques are very labor intensive, and (4) quality assurance techniques and processes are in their formative stages and are time consuming.

[1]Aluminum, titanium, and steel will continue to be important materials in future aircraft structural applications.

[2]Total recurring cost includes recurring engineering, recurring tooling, manufacturing labor, material costs, and quality assurance.

Projections for the mid-1990s indicate that the hours per pound for both conventional metal materials and composite materials will decrease. However, composites will still be on the order of two times more expensive than aluminum (per pound). These projections were based on a company's ongoing development work and capital investment projections. Unfortunately, at the time the worksheets were distributed (1987), the outlook for the defense industry was substantially brighter than the current one. Thus, given current production stretchouts and cancellations, it is not clear that companies will be willing (or able) to make the capital investments necessary to achieve the projections reported here.

In addition to the structural cost factors described above, an approach for estimating overall airframe costs was developed. The method, which is suitable for use in a program's conceptual stage when little detailed design information is available, takes into account not only the cost of the airframe structure but also the cost of the airframe subsystems (e.g., electrical, hydraulic, environmental) and the cost of final assembly/integration. Reduced to its simplest form, the method applies weighted material indexes to baseline cost estimating relationships that utilize aircraft empty weight and maximum speed as the principal explanatory variables.

The method was applied to two hypothetical fighter aircraft to get some feel for the net effect of composites on overall airframe costs. One of the fighters weighed 13,000 lb and had an all-aluminum structure. The other fighter weighed 11,700 lb and was based on the assumption that only 5,200 lb of graphite/epoxy would be required to replace 6,500 lb of aluminum (a 20 percent weight savings). A comparison of the costs of the two vehicles indicated that the substitution of graphite/epoxy for half the aluminum in the structure would increase nonrecurring airframe costs by about 3 percent and recurring airframe costs by roughly 35 percent.[3] However, acquisition cost is by no means the only criterion for deciding whether or not to utilize advanced materials such as composites. Operations and support costs must also be considered and the net life-cycle cost effect then balanced against the performance advantages.

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[3]This conclusion assumes that we started with an all-aluminum aircraft and that the only weight savings was that from the material substitution. Different material mixes as well as potential aircraft resizing would produce different results.

ACKNOWLEDGMENTS

Lieutenant Colonel Terry Luettinger, formerly of the Cost Programs Division of the Office of the Deputy Assistant Secretary for Cost and Economics, provided invaluable sponsorship of this work. John Birkler, a RAND colleague, originally proposed this work and was instrumental in determining the scope of the project. Robert Neff, formerly of the Aeronautical Systems Division, and James Weathersbee, formerly of the Naval Air Systems Command, assisted in the distribution of the worksheets to the major U.S. aircraft and helicopter companies. Arthur Strathman provided crucial insights into the costs of composite materials.

RAND colleague John Matsumura and Fred Timson of Northrop provided timely and thorough reviews. Finally, the following individuals provided the data without which this analysis could never have been undertaken.

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I. INTRODUCTION

Military applications of advanced materials—composites and new metal alloys—are found in aircraft, satellite, and missile structures; engine parts; solid rocket motor casings; and protective armor. This study concentrated on the issues surrounding advanced material usage in military aircraft. Emphasis has been placed on composite materials because of their relative importance in such applications, although advanced metal alloys are also described.

Composites can offer several advantages over conventional materials. Their superior lightness, strength, and stiffness aid in improving aircraft maneuverability and expanding flight envelopes. They also have greater fatigue and corrosion resistance than conventional metals. Weight savings achieved through the use of new materials may increase payload-range capability, provide the opportunity to downsize subsystems at a constant performance level, or allow for better fuel efficiencies. Because composite parts are built up ply by ply, unitized designs, which may require fewer fabrication and assembly hours, are possible. Some composites may also play a role in meeting low-observability requirements because they have desirable electrical properties.[1]

Given the evolving state of materials technology, considerable uncertainty surrounds the acquisition cost of future aircraft structure. Factors that contribute to the uncertainty include:

- The material composition of future aircraft.
- Specific properties of commercially available materials.
- Material price trends.
- The availability and maturity of material processing technologies.
- The level of automation in aircraft design and manufacturing processes.
- The incentives to invest in the capital equipment necessary to automate the manufacturing process and work with certain categories of new materials given fewer new military programs.

The materials of construction as well as the methods of design and production are changing dramatically. As a result, the use of cost estimating relationships (CERs) based solely on historical data of the 1960s and 1970s to project the costs of airframes in the late 1980s and early to mid-1990s is a dubious proposition. Moreover, considerable amounts of design and production cost data for new materials are not likely to be available for some time. Consequently, this study was undertaken with the goal of providing cost analysts with an interim method for assessing the costs of future aircraft systems. More specifically, the objectives of this study were to (1) prepare a technical primer on advanced structural materials and (2) develop a methodology, suitable for use in the early planning stages of a program, for estimating airframe structure acquisition costs.

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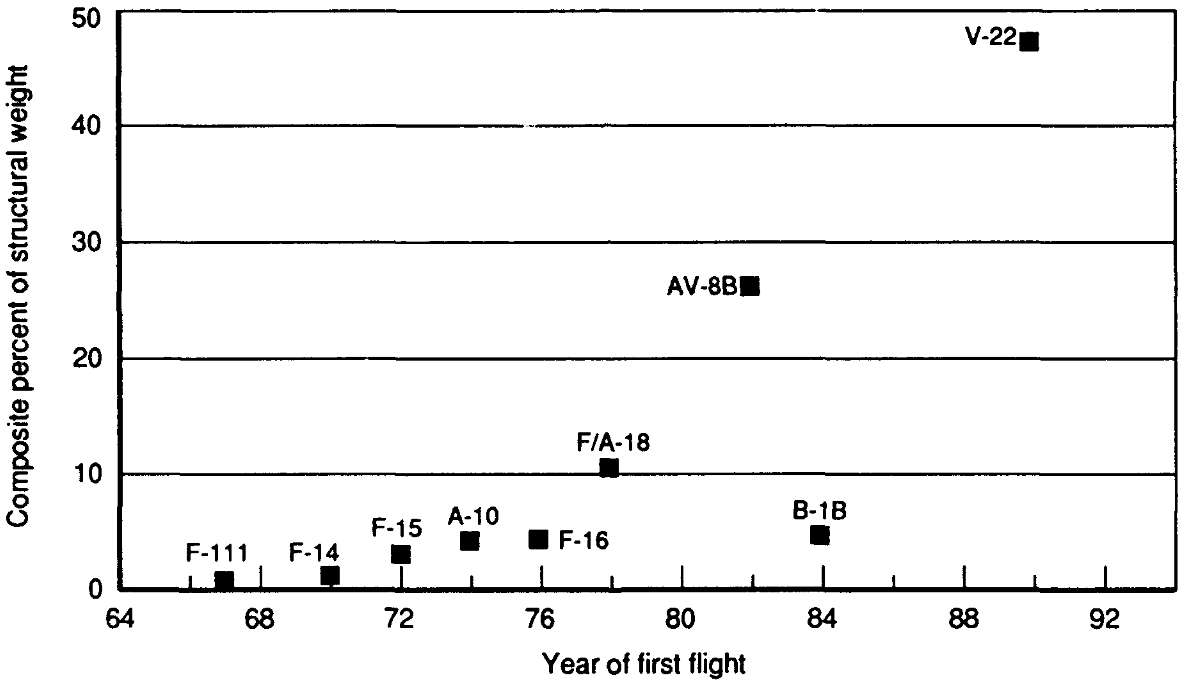
[1]Composites are not without their disadvantages, however. For example, composites are not as damage-tolerant as conventional materials (for low energy impacts such as dropped tools) nor are they as easy to repair. Additionally, because they are a fairly new technology, their costs (on a dollar per pound basis) tend to be higher than costs of more conventional materials.

BACKGROUND

Because of the advantages they offer, advanced composite materials have increasingly been incorporated into aircraft. Figure 1 shows the trend of composite material incorporation into military aircraft as a percent of structural weight. As the figure shows, composites have become an increasingly important material for airframe structural applications.[2]

Initially, composites were incorporated into non-flight-critical secondary structures such as access doors and flaps. Later, as data and experience were gained, the amount applied to aircraft structures increased to the point of being incorporated into primary (load-bearing and flight-critical) structure and complex-shaped components such as wing skins, stabilizer skins, and longerons. For example, boron/epoxy was used in an F-4 rudder and F-111A secondary structure before being introduced into the F-14A horizontal stabilizer (the F-14A was the first aircraft to incorporate composite material in a flight-critical component). Figures 2 through 7 show the structural material distribution on the F-14A, F-15E, F/A-18A, B-1B, AV-8B, and V-22. They are arranged by date of first flight, and they illustrate the range of shapes, sizes, and locations of various airframe subassemblies where composites have been used.



NOTE: In addition to the V-22, several other aircraft now in development are projected to make extensive use of composites—the A-12, ATF (YF-22/YF-23), and the B-2.

Fig. 1—Advanced composite material usage versus first flight date

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[2]The incorporation of composites into commercial aircraft has lagged military applications, primarily because the commercial environment is more sensitive to cost considerations. As points of comparison, only about 1 percent of the Boeing 747's structure weight is composite, while the Boeing 767 has only about 3 percent. However, the Beech Starship, a 4100 lb (structure weight) business aircraft with a top speed of 387 mph, is 67 percent composite (45 percent graphite/epoxy and 22 percent Kevlar and fiberglass).

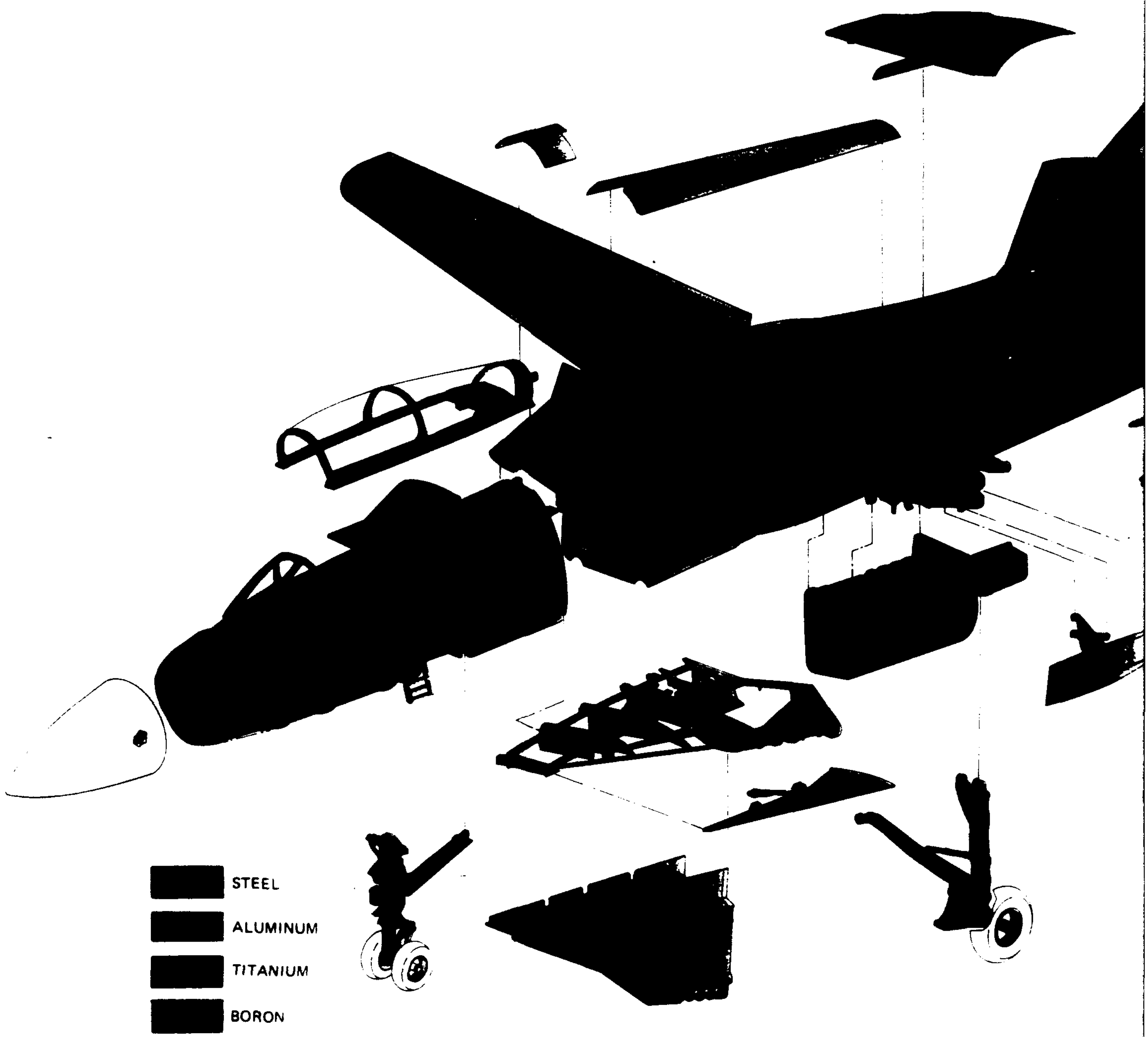

Figure courtesy of the Grumman Corporation

Type	%AMPR	Material	Primary Applications*
Titanium	24.4	Ti-6Al-4V	Electron-Beam Welded Wing Carry-Through Structure, Wing Pi Longerons, Bulkheads, Lower Wing Covers
		Ti-6A1-6V.2Sn	Wing Upper Covers, All Stringers, Bulkheads, Engine Frames, Honeycomb Door Face Sheets
		3Al-2.5V	Tubing
Aluminum	39.4	2024-T81 Sheet & 2024-T851 Plate	Fuselage-Honeycomb, Beaded and Bonded, and Sheet-Stringer Construction; Machined Bulkheads
			Vertical Tail and Rudder
			Wing Ribs, Honeycomb Flaps, Slats, Spoilers and Wing Tip
Steel	17.4	D6AC	Horizontal Tail Support Bulkhead, Main Gear Support Bulkhe:
		300M	Main and Nose Landing Gear
Boron	0.6	Composite	Horizontal Tail Skins

*Remaining 18.2% AMPR is Represented by Purchased Parts, Wiring, Tubing, Etc

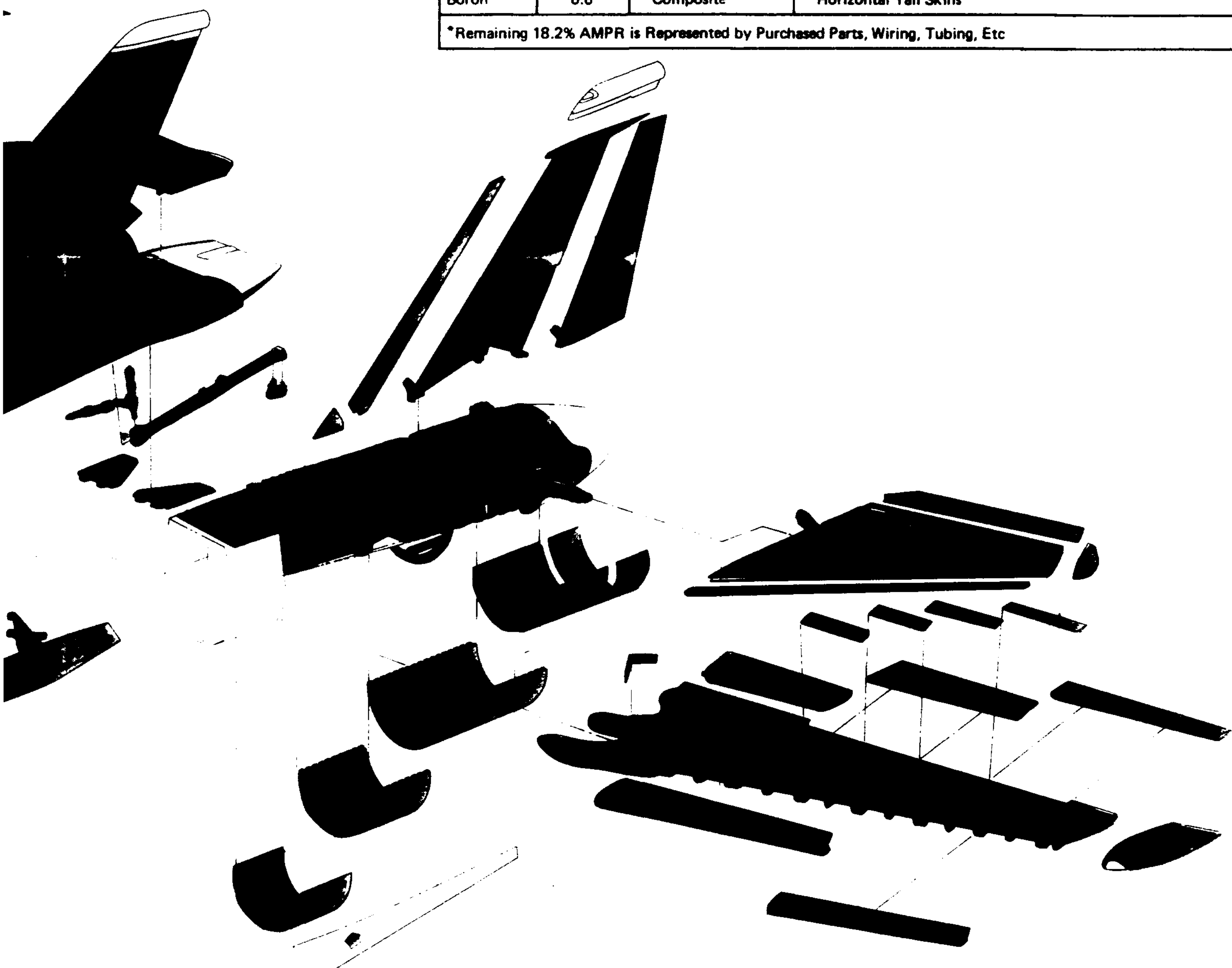

Fig. 2—F-14A structural material distribution

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laterial	Primary Applications*
i-6Al-4V	Electron-Beam Welded Wing Carry-Through Structure, Wing Pivot Longerons, Bulkheads, Lower Wing Covers
i-6A1-6V.2Sn	Wing Upper Covers, All Stringers, Bulkheads, Engine Frames, Honeycomb Door Face Sheets
Al-2.5V	Tubing
024-T81 Sheet & 024-T851 Plate	Fuselage-Honeycomb, Beaded and Bonded, and Sheet-Stringer Construction; Machined Bulkheads Vertical Tail and Rudder Wing Ribs, Honeycomb Flaps, Slats, Spoilers and Wing Tip
6AC	Horizontal Tail Support Bulkhead, Main Gear Support Bulkhead
00M	Main and Nose Landing Gear
omposite	Horizontal Tail Skins
presented by Purchased Parts, Wiring, Tubing, Etc	

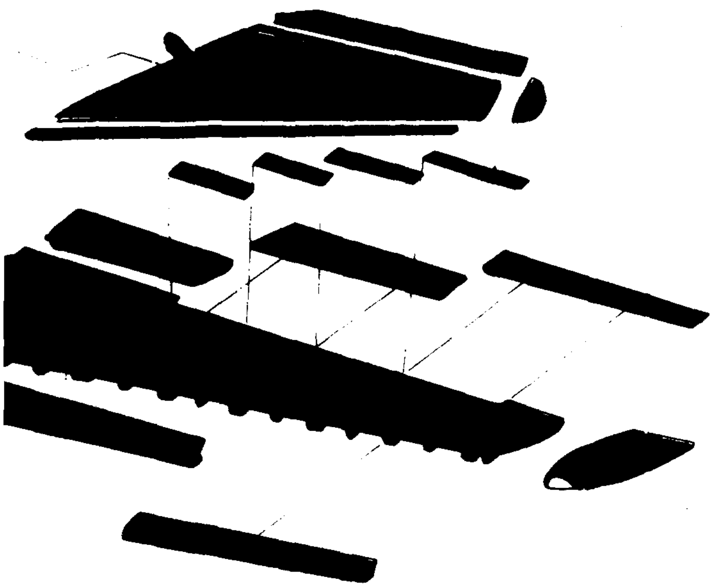

Fig. 2—F-14A structural material distribution

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MANUFACTURING BR
F-15E AIRCRA

Figure courtesy of the McDonnell Douglas Corporation

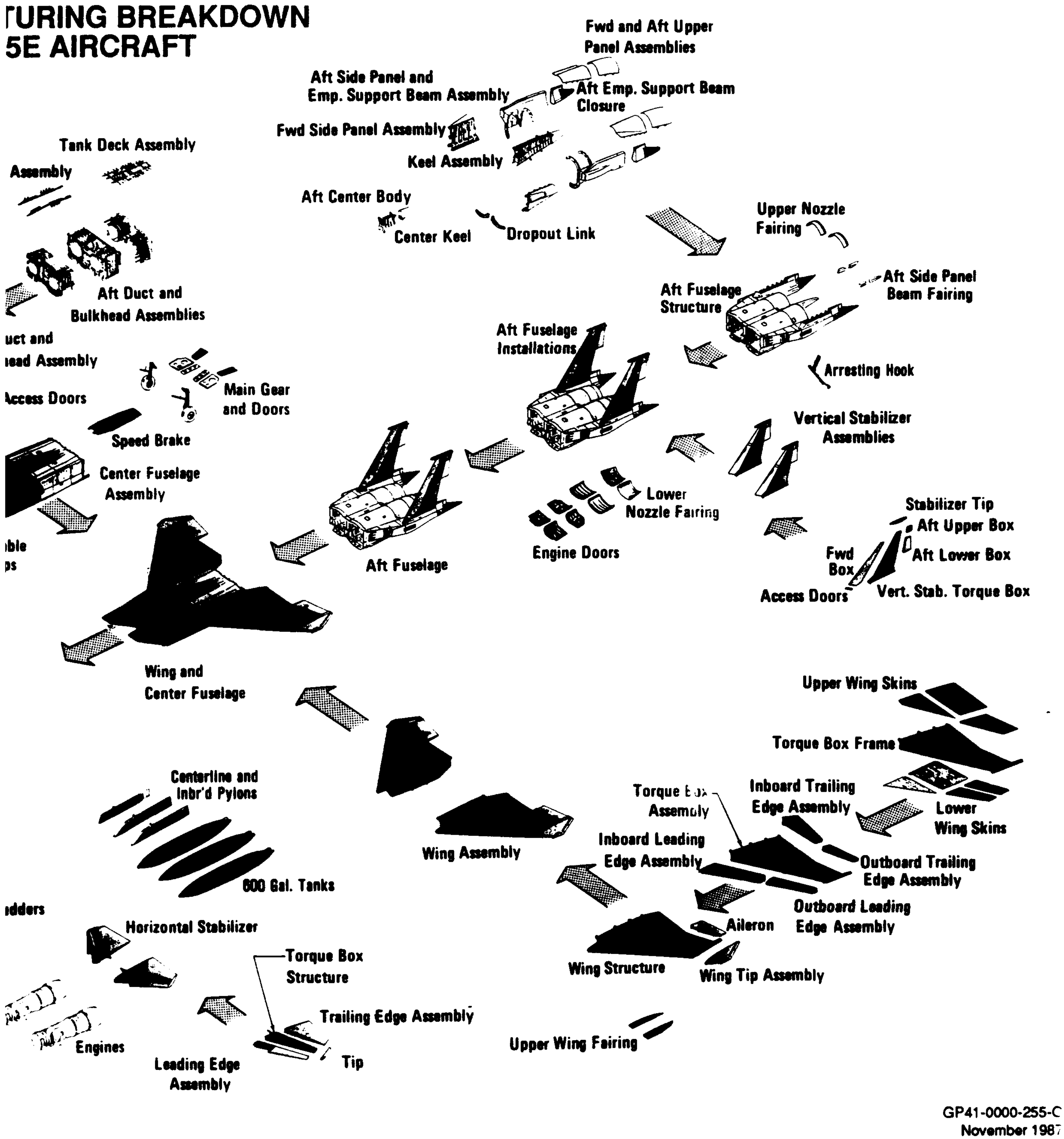

Fig. 3—F-15E structural material distributio

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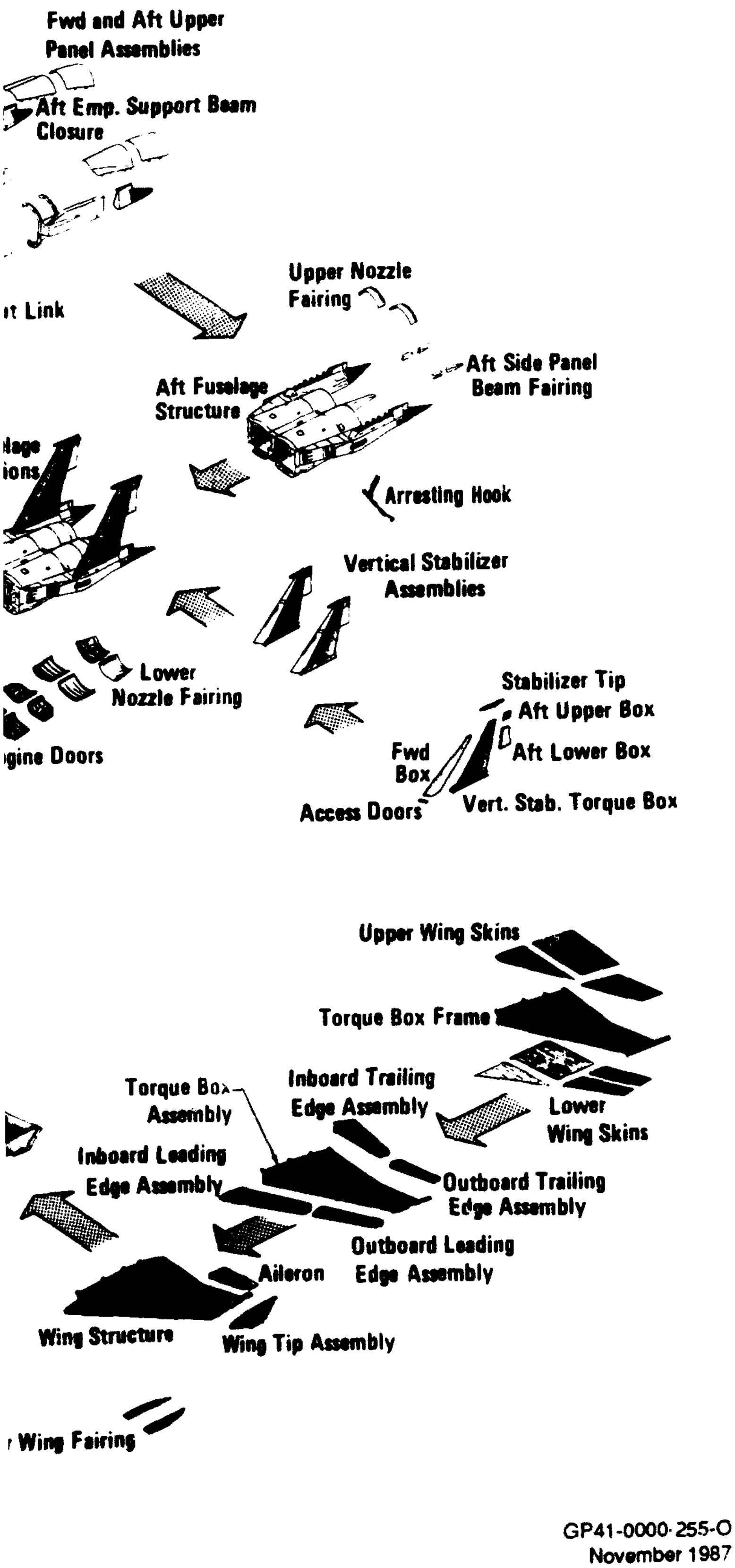

Fig. 3—F-15E structural material distribution

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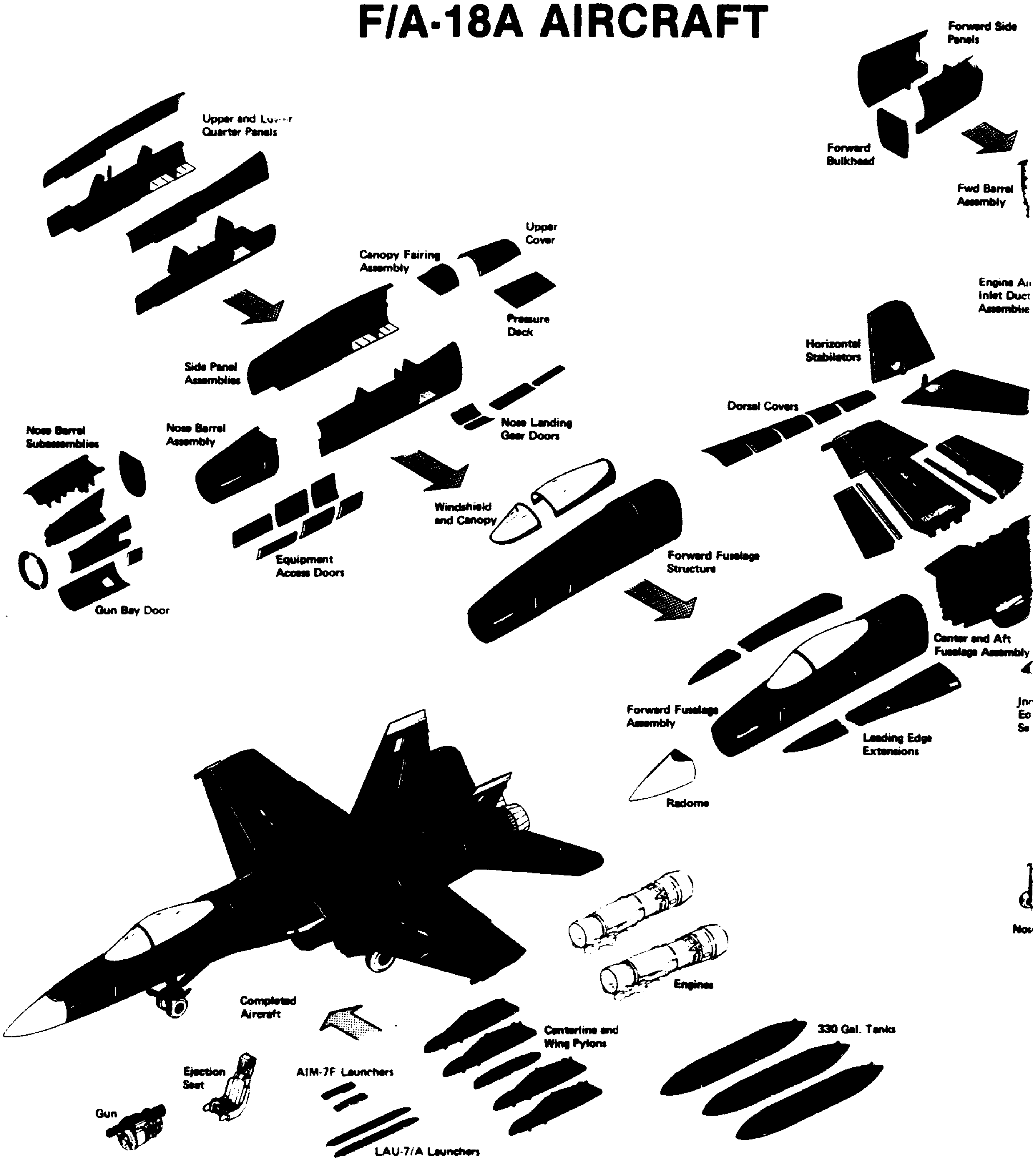
MANUFACTURING BREAKDOWN F/A-18A AIRCRAFT



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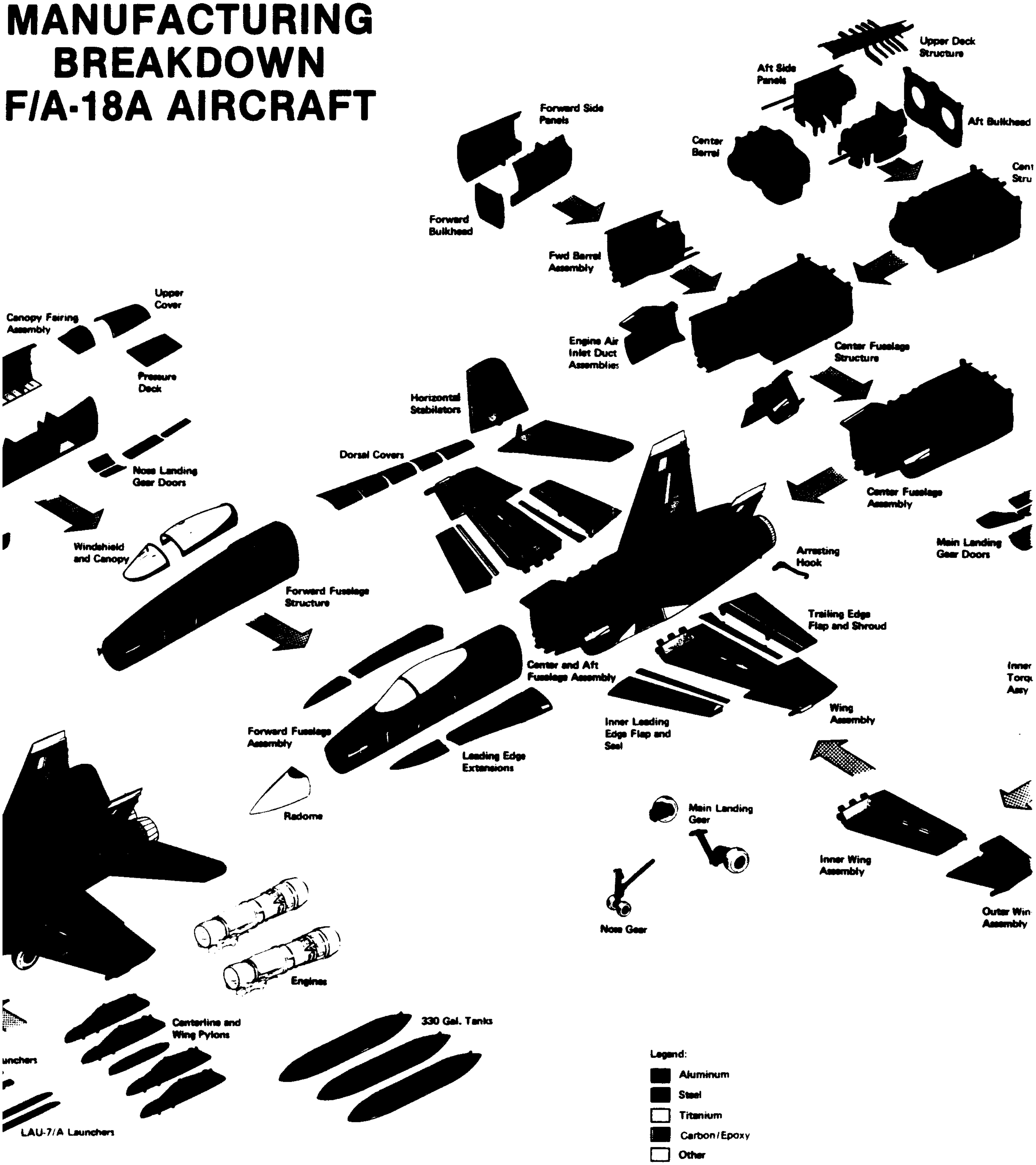
MANUFACTURING BREAKDOWN F/A-18A AIRCRAFT



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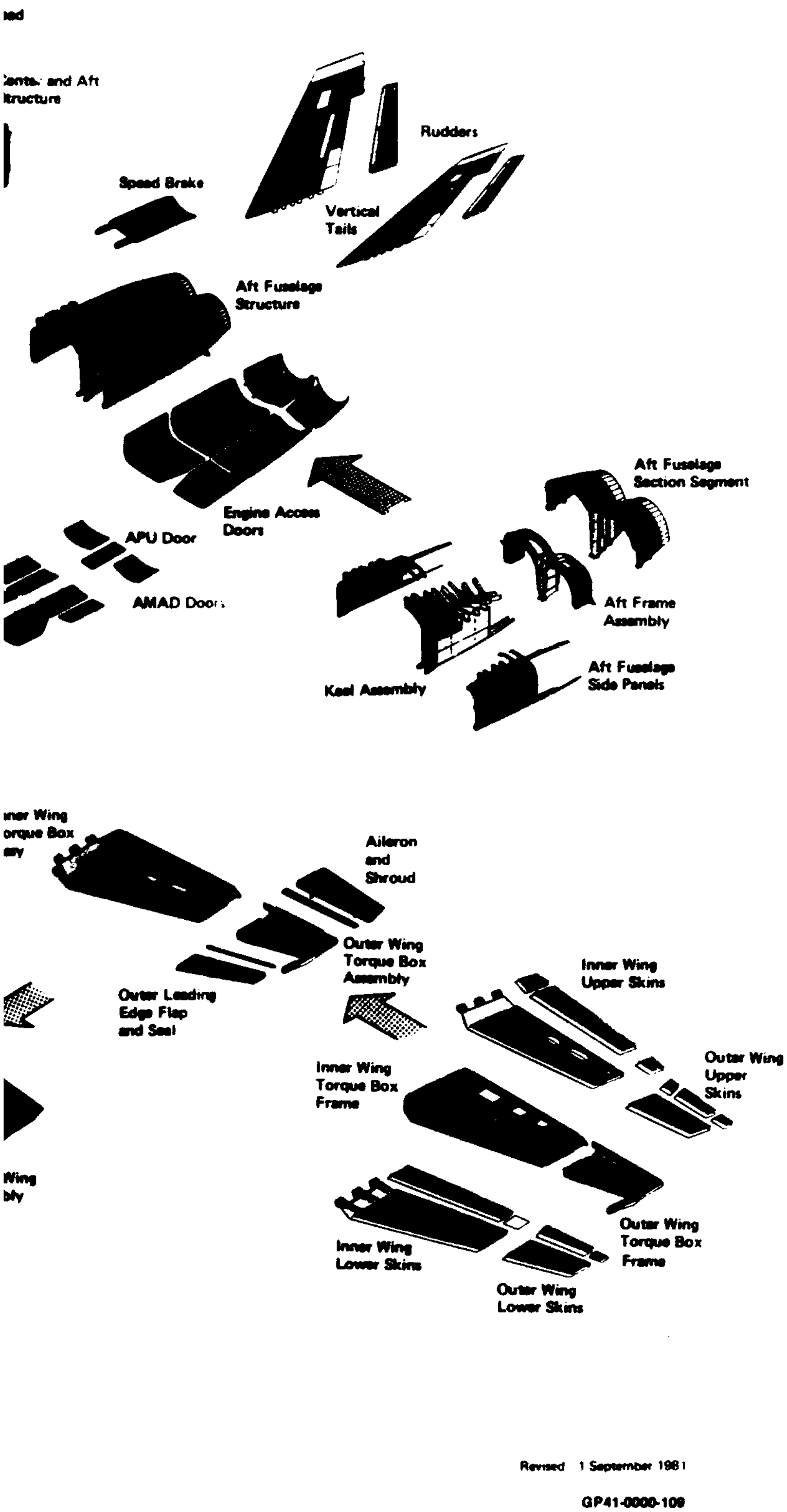


Fig. 4—F/A-18A structural material distribution

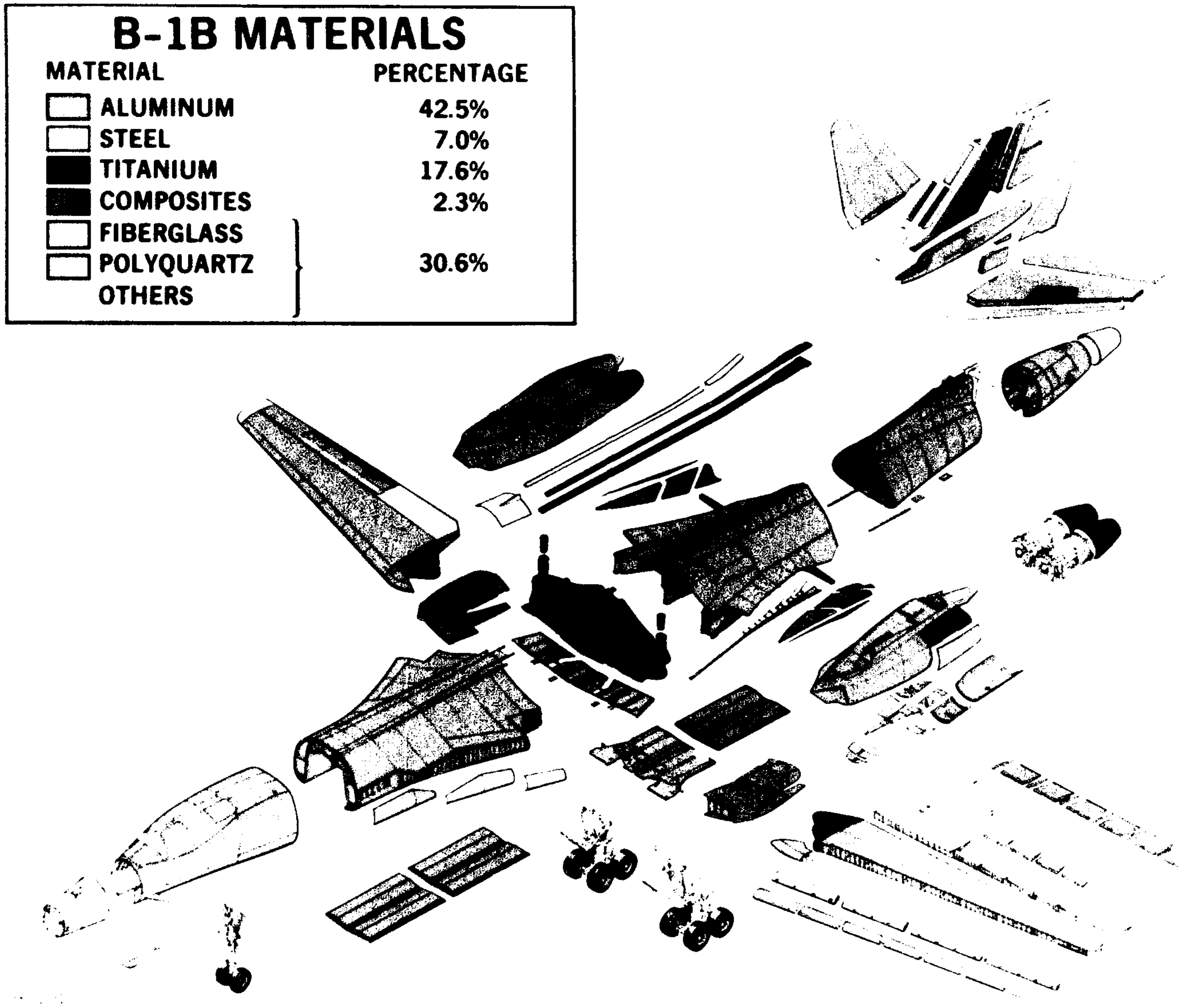


Figure courtesy of Rockwell International

Fig. 5—B-1B structural material distribution

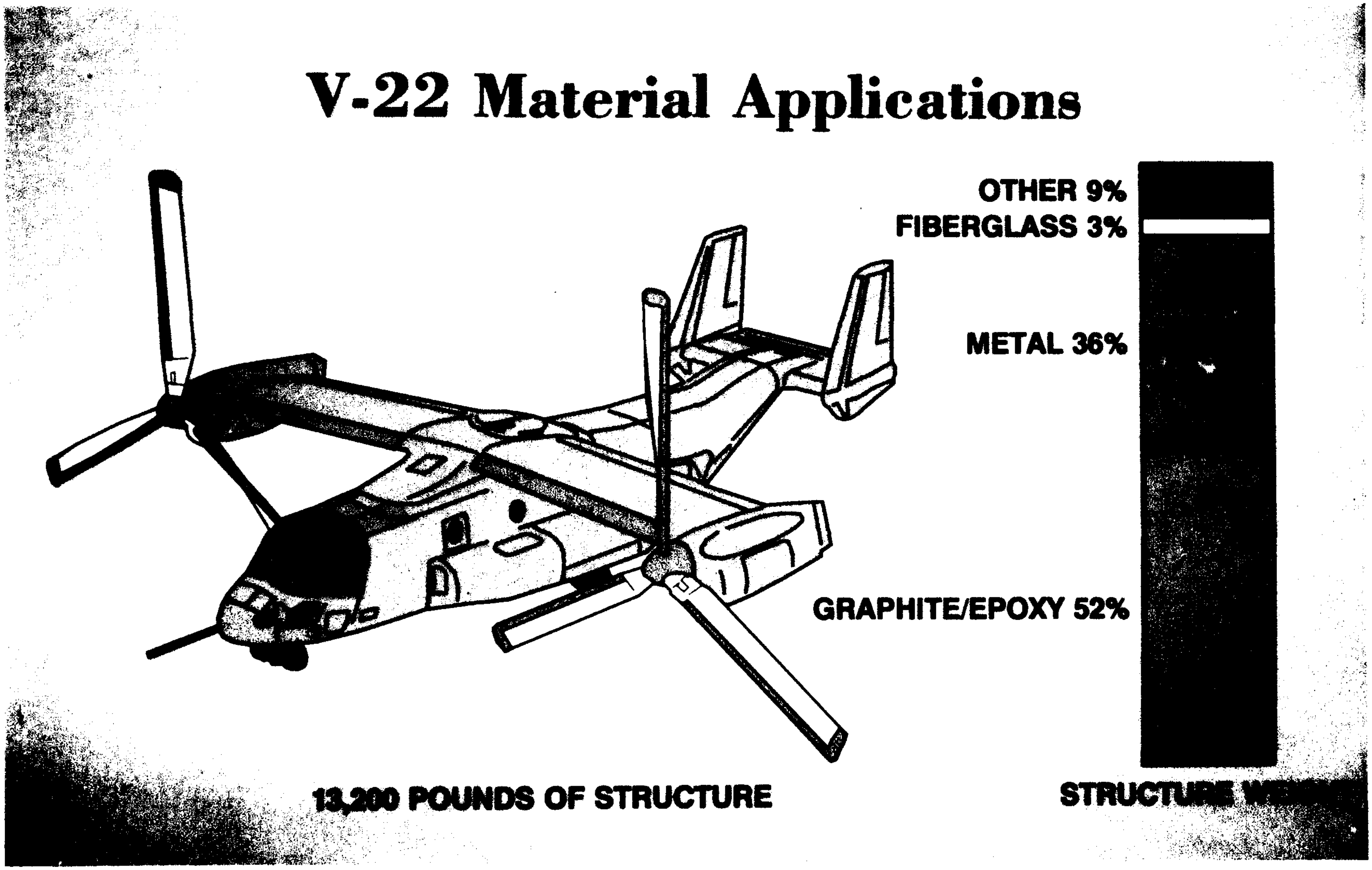


Figure courtesy of Boeing-Vertol

Fig. 7—V-22 structural material distribution

To date, the advanced composite materials of primary importance and usage are the epoxies (Kevlar/epoxy, graphite/epoxy, and boron/epoxy). Greater application of toughened epoxies, graphite/bismaleimide, and other graphite/polyimides is expected in the future. Much attention has also been paid to graphite/thermoplastic materials because of their potential processing ease, although at this time they are much less mature. According to the open literature, both graphite/thermoplastic and graphite/bismaleimide are being planned for the advanced tactical fighter (ATF) prototypes. On the advanced metals side, greater application of aluminum-lithium alloys and superplastically formed aluminum and titanium is expected.

APPROACH

Typically, RAND studies collect actual data and perform statistical analyses to develop CERs.[3] In this instance we believed the many uncertainties surrounding the use of these new materials and the dynamic nature of the material technology and its associated manufacturing processes warranted an industry survey. More specifically, the reasons a survey approach was chosen rather than a statistical analysis of historical data are as follows:

- The current data (for which production experience is available) are limited in terms of:
  - The number of observations. There are only a half dozen historical data points (military aircraft programs) encompassing all composite material types.
  - The range of material types. Some materials, such as aluminum-lithium and graphite/thermoplastic, have not been incorporated into production aircraft; as a result, no historical data, except for data based on developmental experience, exist for these materials.
  - The level of usage. Projected levels of usage are far beyond what has been attained by existing production aircraft.
- The manufacturing technology is rapidly evolving.

The worksheet, a copy of which is provided in App. A, consisted of five sections: corporate history, material usage within an aircraft, material technical information, cost (both nonrecurring and recurring cost elements), and general questions. Information was collected by material type for materials that would potentially be used in airframe structure no later than the 1990s. Materials of interest are those either currently in production aircraft or just recently out of the laboratories.

Two time periods were considered: the late 1980s and the mid-1990s. Data for the late 1980s were to reflect a company's current experience, while data for the mid-1990s were to reflect a company's best judgment regarding the evolution of the basic technical knowledge of the materials as well as of design and manufacturing techniques. Summarized cost information is reported for each of the functional categories listed in Table 1. Improvement curves, weight sizing factors, and material buy-to-fly ratios are also reported.

Although this analysis has emphasized airframe structure costs, overall airframe costs include not only the basic structure, but also airframe subsystems (e.g., electrical, hydraulic, environmental) and final assembly/integration. We have therefore developed a method for estimating *total airframe* costs by combining the results of this study with those of a previous

[3]For example, see Hess and Romanoff, 1987; Birkler, Garfinkle, and Marks, 1982.

Table 1

AIRFRAME FUNCTIONAL CATEGORIES CONSIDERED

Functional Categories	Addressed
Engineering	NR, R
Tooling	NR, R
Manufacturing	R
Quality assurance	R

NR = nonrecurring.
R = recurring.

RAND study.[4] As a result, structure costs are placed in context and not considered in isolation.[5]

Section II contains a description of the new materials for those unfamiliar with composite materials and advanced metal alloys. Section III describes the manufacturing processes associated with advanced materials. The emphasis is on composite fabrication techniques. Section IV presents the nonrecurring and recurring cost information collected with our worksheets, and Sec. V describes our suggested methodology, with an example. The last section, Sec. VI, presents our conclusions. The appendixes contain a sample of the worksheet distributed to industry (App. A), brief synopses of other composites cost studies (App. B), definitions of cost elements (App. C), and definitions of terms (App. D).

[4]Hess and Romanoff, 1987.

[5]To compare this data collection effort and methodological approach with the literature, see App. B, which briefly describes other recent composites cost studies.

II. MATERIAL CHARACTERISTICS

This section discusses new material composition, properties, and general characteristics. The intent is to familiarize cost analysts with the new materials so that they may better understand them and be able to assess the uncertainties and risks involved.

Advanced materials are classified into the following categories:

- Advanced composite materials: ceramic matrix composites, polymer matrix composites, and metal matrix composites.
- New metal alloys: aluminum-lithium and powder metallurgy alloys.

Composite materials, polymer matrix materials in particular, are the new materials of primary concern for future aircraft structure and as a result are emphasized in this report. Their strength, stiffness, and cost properties are particularly suited for aircraft structural applications. Metal and ceramic matrix composites are at a much earlier stage of development than polymer matrix composites. Metal matrix composites offer superior temperature capabilities but parts fabrication is still difficult. Ceramic matrix composites also have superior temperature capabilities but are brittle and therefore not suitable for most structural applications.

Aluminum, titanium, and steel, the materials historically used in aircraft, will continue to be used. Titanium is used in military aircraft where strength, toughness, heat resistance, and high structural efficiency are required. The Air Force Materials Laboratory is studying powder metallurgy alloys, but they are still in a developmental stage and their incorporation into aircraft is further on the horizon. Consequently, they will not be discussed here in detail.

Composite materials have two constituents, the fiber and the matrix. It is common practice to refer to a composite material first by fiber (or reinforcement) material type and then by the matrix type (see Fig. 8). The matrix holds the fibers together, redistributes the loads between the fibers, and plays an important role in many of the operating characteristics of the composite material (service temperature, resistance to chemicals, damage tolerance, etc.). The fiber/matrix interface can also be an important consideration since relative slippage or disbonding can occur under various conditions. The fiber gives the material its strength and stiffness properties and is thus the load bearing element. Fibers may be short (chopped) or long (continuous). The five fibers predominantly used are glass, Aramid (DuPont's Kevlar is in this class), graphite (carbon), quartz (silica), and boron. Glass reinforced plastic, or fiberglass, was first used in the 1940s and 1950s in small boats, aircraft radomes, and automobiles. It was the first composite material developed, but it is not considered an advanced composite because of its low-value properties. Carbon fiber, developed in the 1960s, is stronger and stiffer than glass, making it a primary candidate for structures with high loads. Aramid fibers lie somewhere in between glass and graphite fibers in terms of strength and stiffness. Aramid's toughness makes it suitable for ballistic protection.

COMPOSITE MATERIALS

Reinforcing Elements (Fibers)

The reinforcing elements provide the required stiffness and strength necessary to carry the primary structural loads. There are three forms of reinforcements: continuous fibers, short or long whiskers, and particulates. Continuous fibers may come in unidirectional tape or

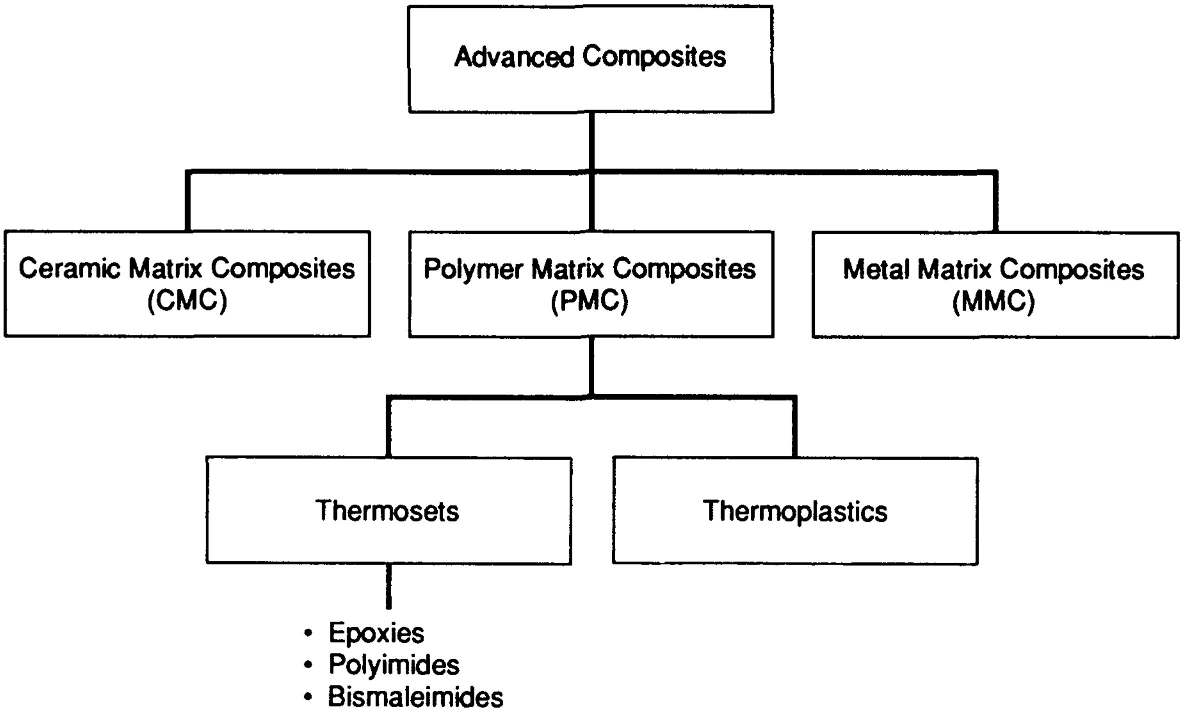
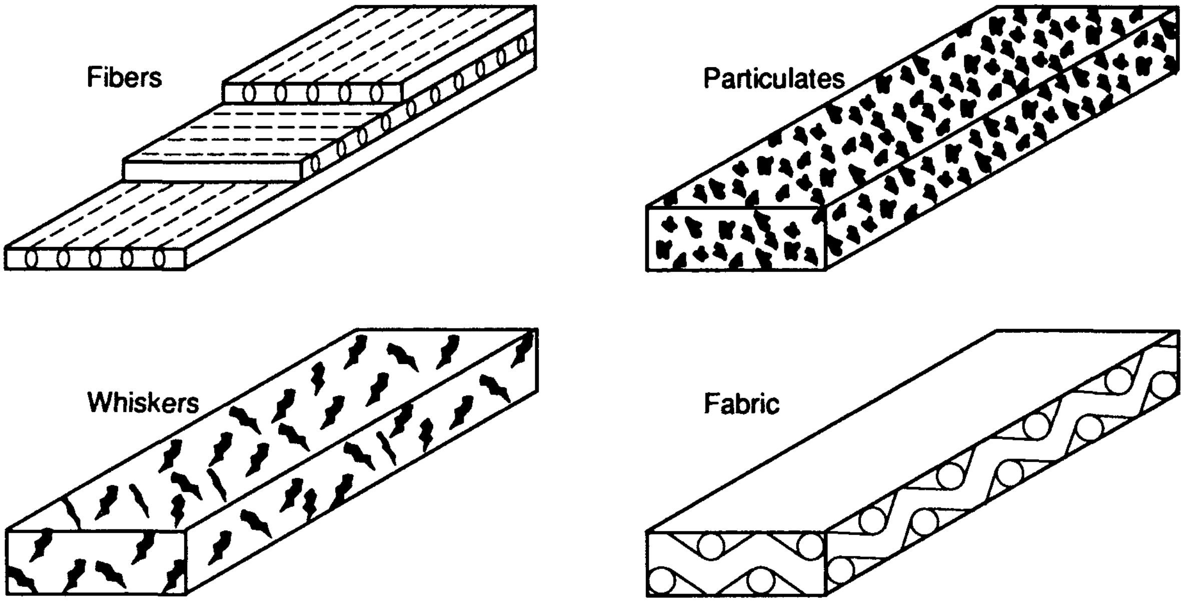


Fig. 8—Advanced composite classifications (by matrix material)

woven (fabric) form. The various forms are shown in Fig. 9. Fiber reinforcements dominate military applications and are the only reinforcements considered in this report. The effectiveness of any reinforcement is largely determined by volume fraction, orientation, and length. The volume fraction of the reinforcement is the percentage of the total volume of the part filled with the reinforcing material. Thus, a volume fraction of 0.6, typical of many graphite/epoxies, means that parts fabricated from these materials are composed of 60 percent fibers and 40 percent matrix material (by volume).

In addition to such factors as the stiffness, strength, and the volume fraction of the fibers, the effect of fiber orientations on final part properties is also very important. The concept here is the relative alignment between the directions of the critical loads and the directions of the fibers; the greater the alignment between the fibers and the loads, the greater the structural efficiency. As the fibers become more randomly distributed with respect to the applied loads, structural efficiency reduces substantially. Thus, randomly oriented fibers are not used in primary structures. Designers will strive to align the fibers and the loads to minimize the laminate thickness (and the weight of the part). However, in many practical parts, a certain degree of structural efficiency (weight) is traded for higher margins of safety, which accounts for the uncertainties associated with composites.

The optimum ply layup angles (or fiber orientations) for a complex composite part are not completely determined in advance. During the design process, the orientation of the fibers in a part may experience several iterative changes, for several reasons: a better understanding of the loads that will be applied to the structure, the incorporation of producibility concerns,



SOURCE: New Structural Materials Technologies: Opportunities for the Use of Advanced Ceramics and Composites, OTA, September 1986, p. 43.

Fig. 9—Reinforcement forms

changes in the stress analysis, etc. Also, the relative newness and uncertainty associated with composites can also result in conservatism with respect to the laminate thickness and the layup angles.

Finally, as the length-to-diameter ratio of the fibers increases, the reinforcing effect increases. Therefore, long, continuous fibers contribute more stiffness than short fibers.

Graphite filaments are composed of thousands of individual fibers. The diameter of a single graphite fiber is about 0.0004 inches. Filaments are packaged in a variety of ways in order to suit the needs of industry. Unidirectional tape and multidirectional fabrics are most common. Unidirectional tape has the filaments aligned with the rolling direction of the tape. Fabrics are supplied with filaments running in two (or more) directions. Often, these fabrics and tapes have been preimpregnated with a resin matrix material and are called "prepreg" tapes. Although fibers can be woven together in three or more directions, the maximum weight savings will generally be achieved when uniaxial tapes or biaxial fabrics are used; many structures are loaded predominantly in one direction, and a weight penalty must be paid if extra fibers are aligned at angles to the load.

For most airframe applications, continuous graphite (carbon) fibers have been used as the reinforcing elements. Such other materials as boron and Kevlar (Aramid) are also used, but carbon fiber technology has dominated because it provides fairly high performance at "reasonable" cost. Boron fibers surpass graphite in terms of strength, but the cost of boron (raw material) is about three times that of graphite. Boron fibers are also more difficult (and expensive) to machine because they are harder and about ten times thicker than graphite. Such thick fibers are limited to parts having radii of curvature greater than 3 in. and thus are eliminated from many substructural applications (graphite fibers may be used in parts with

radii of curvature as small as 1/16 in.).[1] Kevlar fibers have very good tensile strength properties and are cheaper than graphite. However, they are considerably less stiff and have lower compressive strength. These weaknesses eliminate Kevlar from many critical applications.

Two measures of performance potential, "specific stiffness" and "specific strength," are defined by dividing the stiffness (or modulus) and strength of the material by its density. The results are a measure of a candidate material's efficiency and are a useful comparative tool in an industry that often describes structural components as "stiffness critical" or "strength critical." Although there are other issues to consider, a comparison of these normalized properties highlights the superior performance potential of the advanced composites (see Figs. 10 and 11). On a "per pound" basis, the advanced composites are inherently stiffer and stronger than metals.

The properties in Figs. 10 and 11 are given for both compression and tension to highlight the fact that some materials (Kevlar, boron, glass) perform very differently depending on their stress state.[2] Usually, the compressive strength of the material is the weaker of the two; this is

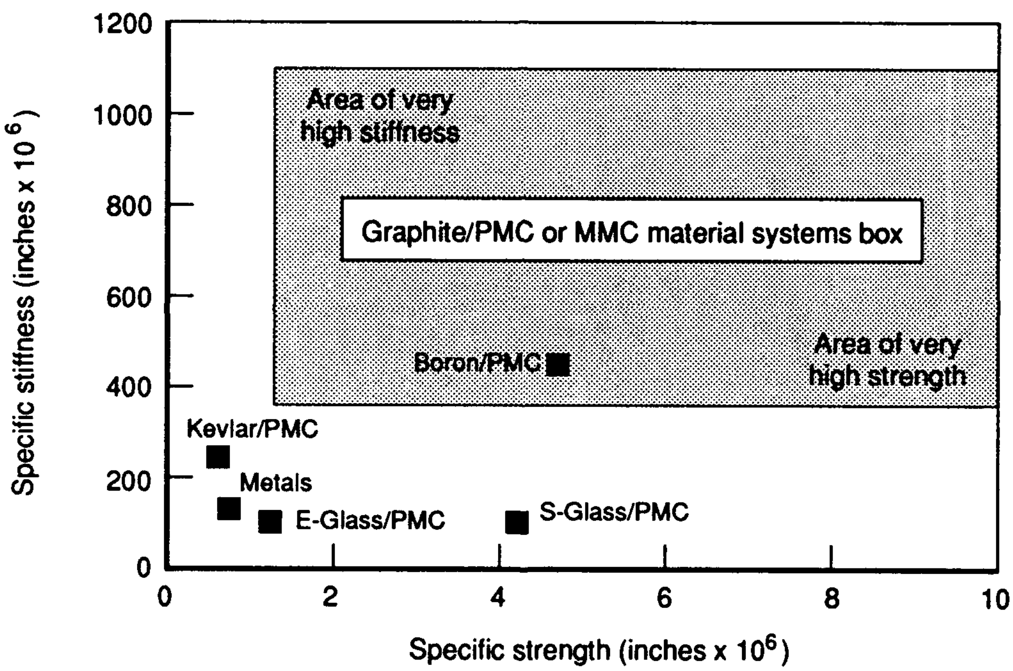


Fig. 10—Compression comparison: Specific stiffness vs. specific strength for unidirectional prepreg

[1]*DoD/NASA Advanced Composites Design Guide*, 1983.

[2]The distinction between simple tensile and compressive stress states can be made in the following way. If an object is undergoing relative elongation in the direction of the applied load and a simultaneous "thinning" in the transverse (or no-load) direction, it is in a state of tension. If the object is shortening in the direction of the load and "thickening" in the no-load direction, it is in compression. A simple but extreme example of this distinction can be made with putty. Pulling on a piece of soft putty from both ends produces a very noticeable elongation and thinning to the point of breakage. Squeezing the putty results in a shortening and thickening due to compression. Compressive loads may result in a buckling instability long before the ultimate compressive strength of the material is reached. The

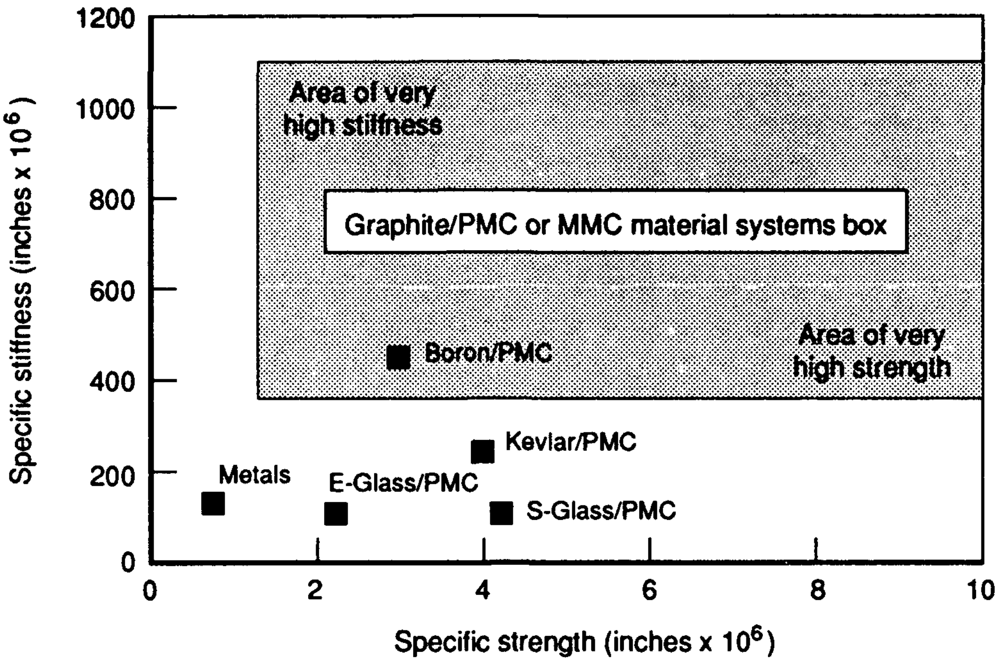


Fig. 11—Tension comparison: Specific stiffness vs. specific strength for unidirectional prepreg

true for Kevlar and glass. However, boron fibers are actually stronger in compression than in tension. Graphite fibers have approximately equal properties in tension and compression. Such responses must be considered when deciding which materials are appropriate for certain parts. Kevlar is very common in pressure vessel applications because the fibers are predominantly in tension, and Kevlar is fairly cheap and damage tolerant. Graphite is more appropriate for structures that are subject to substantial bending loads, which generate areas of both tension and compression in the part. In Figs. 10 and 11, the rather large boxes representing graphite/epoxy reflect the general range of properties that can be provided by suppliers, depending on the specific application. Note that not every point in the graphite/epoxy boxes can be achieved in practice; they reflect the approximate range the materials' properties currently cover.

Certain properties of fibers can be enhanced, but often that will be at the expense of other properties. For example, graphite fiber stiffness and strength are inversely related; as one is enhanced, the other is decreased. Several varieties of graphite fibers are on the market, ranging from high strength graphite (with lower stiffness) to high stiffness graphite (with lower strength).

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predisposition of a structure to buckling instabilities can be mathematically calculated and is a strong function of the structure's geometry as well as its material stiffness. For example, a long, thin steel rod may buckle at low compressive loads. A short, thick steel rod will probably never buckle and will simply fail (crush) when the compressive strength of the steel is reached.

In summary:

- The fibers provide structural stiffness and strength (which can save weight).
- Graphite fibers have the best combination of strength, stiffness, and cost and are therefore the most widely used fiber.
- The material formability and machining characteristics are largely determined by the fiber selection.
- Fibers are provided in many formats: unidirectional (tape), bidirectional (fabric), etc.
- Final part properties are largely determined by fiber volume fraction, fiber orientation, and fiber length.

Matrix Materials

The function of the matrix is to facilitate load transfer between the fibers, which plays an important role in the longevity and damage tolerance of a composite part. Damaged fibers must be able to redistribute their loads through the matrix to the surrounding undamaged fibers.

An intuitive example of the role of the matrix is the response of a composite part to compressive loads. Although the fibers resist most of the load, the matrix lends the necessary geometric support. If the matrix disintegrates or dissolves, the fibers cannot carry any compressive loads and the structure collapses. Thus, the condition of the matrix is extremely important with respect to compressive structural loads. Tensile loads, however, may still be carried by the fibers even if the matrix fails.

The matrix/fiber interface may also help to stop cracks that originate in the fiber from propagating catastrophically. A local impact can break a few fibers, but crack propagation should be inhibited by a ductile matrix or a fiber/matrix debond.

Matrix mechanical properties such as stiffness and strength are substantially lower than those of the fibers. Because of this relative weakness, the matrix determines the mechanical, thermal, and chemical limits and the relative moisture levels of the material system. In operational terms, the matrix determines the environmental limits (temperature, chemical solvent exposure, humidity, etc.) of the composite part.

Industry has been working to develop tougher and more damage resistant matrix systems, but such advances have occurred more slowly than fiber stiffness and strength improvements. Also, superior performance requirements often translate into additional processing difficulties and hence costs increase (at least until the manufacturing processes become more mature).

The advanced composite material systems can be divided into three general categories: polymer matrix composites (PMC), metal matrix composites (MMC), and ceramic matrix composites (CMC). PMCs have prevailed in the airframe industry for more than 25 years because they strike an effective balance among performance, versatility, and cost. Metal matrix materials have superior high temperature properties but are very expensive to manufacture. Although several demonstration components have been produced from metal matrix materials, the demand for them is held low by cost considerations. The mechanical properties of ceramic matrix composites are unsurpassed at elevated temperatures, but they are practically unused in airframe applications because of very low fracture toughness values. Current research emphasizes the toughening of ceramic matrix composites. Each matrix type is discussed below in more detail.

Polymer Matrix Composites. Polymer matrices can be classified as "thermosets" or "thermoplastics," depending upon their chemistry and the cycle that is required to process

them. Both have organic chemistries and service temperature limitations between 200 and 600°F.

Thermosets have polymer chains that cross-link during the application of a prescribed schedule of temperature and pressure (cure cycle). This structure promotes dimensional stability, solvent resistance, and high temperature resistance, but it is also irreversible and slow to cure. For example, a typical epoxy part will require about 8 to 12 hours of processing time, and once cured, it cannot be reprocessed. Most epoxies require processing temperatures around 350°F.

The defense industry has been working with fiber-reinforced, thermosetting epoxies for 25 years and is thus fairly well aware of their strengths and weaknesses. Concern about the limited service temperatures (275°F under hot/wet conditions) and low toughness of epoxies resulted in the development of bismaleimide and polyimide systems. Bismaleimides are actually "addition" polyimides that have lower processing (and operating temperatures) than the rest of the polyimide family. Bismaleimide service temperatures can reach 450°F and can be processed with the same equipment (autoclaves) used for epoxies. Other polyimides can reach service temperatures of up to 600°F, but their processing temperature exceeds 650°F. This is an important manufacturing threshold; major modifications to manufacturing processes and equipment are required above 625–650°F. Thus, polyimide parts are more expensive to produce. Production parts made from these advanced thermosets have begun to appear (e.g., graphite/bismaleimide parts on the AV-8B).

Epoxies are hampered by several concerns. Perhaps the primary concern is low toughness. This means that extensive internal damage (delamination), invisible to the naked eye, may result from such low energy impacts as the dropping of a tool. This damage can have a severe degrading effect on the compressive properties of composite parts. Consequently, composites are penalized by the use of low design strains in order to conservatively compensate for this weakness. Tougher matrices would allow designers to increase the composite design strains and have higher confidence.

Over the lifetime of a part, some epoxies gain weight because of their tendency to absorb water. Obviously, this undesirable feature reduces mechanical properties and vehicle performance factors. Additionally, vehicles that engage in prolonged supersonic or hypersonic flight will not be able to use epoxies (because of a 275°F service temperature limit) in areas of extreme aerodynamic heating. In such areas, the alternatives might include polyimides, metal matrix materials, and carbon/carbon.

Thermoplastics have been the subject of intense research because of their potential for increased toughness, short processing times, and reformability of parts. In 1982, polyetheretherketone (PEEK), was the first thermoplastic to have both reasonable high temperature performance and good solvent resistance. Until then, either of the two, but not both, of those characteristics could be achieved in a thermoplastic. Both, however, are necessary in most airframe applications. There are many thermoplastics on the market; some of the more common include polyetheretherketone (PEEK), polyphenylene sulfide (PPS), and polyetherketone (PEK).

Thermoplastics do not undergo any permanent chemical transformations (such as crosslinking) during the fabrication process. This means that scrap and mishandled parts might be reformed by the application of heat and pressure. Thermoplastics are more flexible than thermosets with respect to cure cycle perturbations; although small changes do not seem to bother thermoplastics, they can have substantial effects on thermoset parts. In addition, thermoplastic prepregs do not require refrigeration, and, perhaps most important, part

fabrication times are measured in minutes rather than hours (thermosets). Thermoplastics can also be welded.

Thermoplastics are potentially more resistant to impact damage and cracking than thermosets. This could have a considerable effect on design strain levels and, therefore, weight savings. High temperature strength and solvent resistance are still fairly low. These deficiencies are being examined and there is evidence to suggest that improvements will be made to make them more competitive with thermosets. The long-term mechanical properties (fatigue and creep) of reinforced thermoplastics are uncharacterized. Fatigue and creep tests are expensive and time consuming to perform; they are not often done in the early stages of material development. Confidence in the material is therefore reduced until these properties can be well characterized.

Other deficiencies of thermoplastics include poor drapability and lack of tack. This means that the prepreg plies are stiff, difficult to form, and do not tend to stick together after pressure has been applied. Heat must be applied to thermoplastic prepregs during the layup process in order to enhance ply formability. Therefore, some modifications will have to be made to any automated layup procedures that have been designed for thermoset plies (which are naturally formable and sticky).

High processing temperatures (650–800°F for thermoplastics versus 350°F for thermosets) are also a serious consideration. When processing temperatures exceed 650°F many conventional manufacturing procedures and equipment must be revised or replaced. For example, metal tools will degrade very quickly, rubber tools cannot be used at all, and autoclaves are subject to extreme wear. Manufacturing processes that minimize autoclave use for thermoplastics are being developed (such as pultrusion).

In Fig. 12, thermoplastic and thermoset processing temperatures and times are compared. An autoclave cure cycle for a thermoset part can require as long as 12 hours when heat-up and cool-down times are considered,[3] but many thermoplastic parts will probably require less than one hour of processing time. However, part consolidation for thermoplastics requires temperatures in excess of 650°F. Such temperatures place heavy demands on both autoclaves and the conventional manufacturing techniques, which have been oriented toward the mild processing temperature (350°F) of thermosets. In fact, autoclave use tends to negate the advantages of thermoplastics because a substantial amount of time is required just to heat up (and cool down) all of the air and mold mass within the autoclave to such high temperatures. Therefore, other fabrication concepts are required and are being developed. However, autoclave use may be unavoidable for certain kinds of parts. According to a National Research Council report,[4] the following processes have been used to either consolidate or form continuous fiber reinforced thermoplastic parts:

- Autoclave lamination and molding.
- Continuous lamination.
- Filament winding.
- Pressure forming.
- Pultrusion.
- Roll forming.
- Vacuum forming.

[3]This includes autoclave time only. Post curing is generally done in less expensive ovens.

[4]National Research Council, 1987.

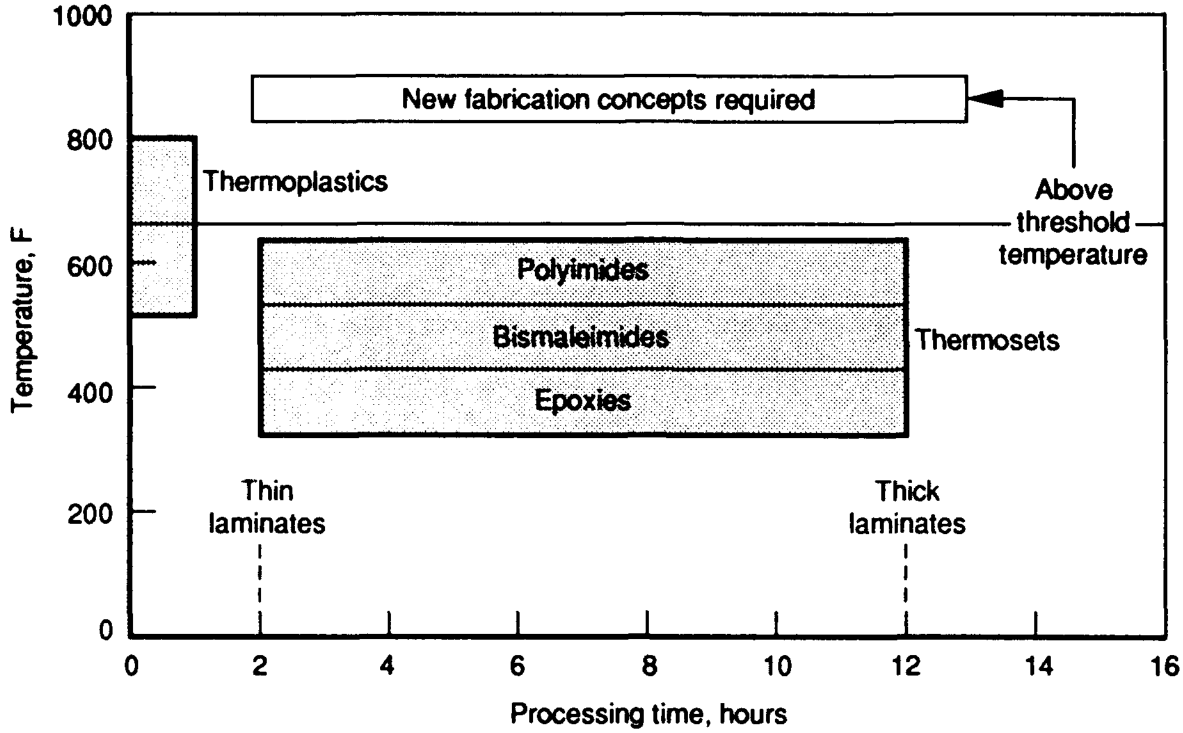

Fig. 12—Matrix processing requirements: Thermosets vs. thermoplastics

Unreinforced thermoplastics as well as those reinforced with short, randomly oriented fibers have been used in low-temperature, secondary structural applications for several years. But advanced composite thermoplastics with continuous fibers are a new concept with little operational experience. Much emphasis is being placed on these materials in current design concepts, but the database is small when compared with fiber-reinforced thermoset systems (e.g., epoxies). It will take time to realize the full advantage of the benefits offered by reinforced thermoplastics and to overcome the technical problems that will surely arise. Some graphite/PEEK wing skin panels for the F/A-18 were installed and evaluated in 1988. Other demonstration components (a torque box, an ATF prototype wing, and a helicopter tail) have been made as well, but long-term, in-service observations have not yet been made. Current thermoplastics should retain their integrity up to 300–350°F, which is comparable to thermoset temperatures.

Metal Matrix Composites (MMC). Metal matrix composites are at a much earlier stage of development than PMCs. These materials offer superior operating temperatures (e.g. 1,000°F for titanium) but are hampered by complex and immature fabrication procedures. Using fiber reinforcements with metal matrices yields superior temperature capabilities while maintaining strength-to-density properties greater than those achieved by superalloys. Metal matrix composites offer better compression strength than polymer or ceramic matrix composites. Aluminum is the most common metal matrix; but other metals are also used, including aluminum, titanium, magnesium, copper, and superalloys. Some of the reinforcing materials include graphite, boron, silicon carbide, and aluminum oxide. Manufacturing processes include

hot pressing, casting, hot isostatic pressing (HIP), pultrusion, forging, and superplastic forming (SPF)/diffusion bonding (DB). Temperatures for these processes can exceed 2000°F, and pressures of up to 2000 psi may be required.

The ATF may offer the first opportunity to test the feasibility of primary MMC structures in a production environment. Lockheed is in the early testing phase of a program intended to demonstrate the superior performance and cost competitiveness of MMC aircraft parts. They are fabricating and testing four vertical tail structural boxes that could be used in an ATF design. These structures have skins fabricated from silicon carbide reinforcements and an aluminum matrix. Previous Lockheed trade studies concluded that MMC fighter aircraft components can reduce structural weight 15 to 30 percent and reduce production costs 5 to 50 percent. Testing of F-15 tail sections made out of metal matrix composites is also being carried out.

Most current applications for MMCs are in aircraft engines and missile structures. A McDonnell-Douglas study concluded that existing MMC materials could greatly improve the range (9 percent) and reduce the structural weight (20 percent) for certain missile designs. MMCs have also found applications in certain space structures (space telescope) where a zero coefficient of thermal expansion is desirable. They may also be necessary for future vehicles that travel for long periods at supersonic speeds. MMCs are also experiencing some use in the auto industry; Toyota and Mitsubishi are producing pressure-cast, aluminum matrix engine parts.

MMCs are currently limited by their high manufacturing cost, fiber/matrix interaction problems, and limited database.

Ceramic Matrix Composites (CMC). Ceramic matrix materials are at an even earlier stage of development than MMCs. These materials are neither organic nor metal. They have superior wear resistance, high temperature strength, and chemical stability compared with metals. Two obstacles to widespread use of ceramic matrix composites are (1) the development of high strength, high modulus, small diameter, continuous fibers whose mechanical properties are not drastically degraded by ceramic matrix processing or operating conditions and (2) fabrication processes that result in a uniform microstructure of nondegraded aligned fiber surrounded by low porosity matrices.[5] Poor fracture toughness properties make them susceptible to sudden, perhaps catastrophic, failure. This drawback removes ceramics from most primary structural applications. Research is being conducted to improve their fracture toughness, and demand for them will increase. Matrix materials for CMCs include carbon, glass, silicon carbide, and alumina. Manufacturing processes include sintering, hot isostatic pressing, hot pressing, melt infiltration, and chemical vapor deposition.

Military applications for ceramics are currently limited to such structures as radomes, bearings, and turbine engine parts. Carbon/carbon is more advanced than other CMCs and is finding uses in such high temperature environments as solid rocket motors and space shuttle leading edges.

Matrix Summary. The primary role of the matrix is to transfer loads between the fibers. It determines the following important properties of composites: resistance to crack propagation (fracture toughness), damage tolerance, solvent resistance, processing temperatures, processing times, and corrosion resistance.

[5]Department of the Interior, 1985.

Thermosets

- Thermoset epoxies have dominated the airframe market.
- Thermoset advantages include good solvent resistance, low processing temperatures and pressures, prepreg formability, and large experience and mechanical property databases.
- Thermoset disadvantages include long processing times, low toughness, high absorption of water, short prepreg shelf life, permanent cure (parts cannot be reformed), and difficulty to repair.

Thermoplastics

- Fiber reinforced thermoplastics may greatly reduce manufacturing costs. Many demonstration components have been made but production run data is not yet available.
- Thermoplastic advantages include very fast part consolidation, reformability, high toughness, and long shelf life.
- Thermoplastic disadvantages include small experience and mechanical databases, high raw material costs, high processing temperatures, poor drapability, and low tack.

Others

- Metal matrix materials have excellent high temperature properties but are still extremely expensive to manufacture and are subject to wide variations in properties between batches. Demands for these materials will increase.
- Ceramic matrix materials are still too brittle for current airframe applications.

Composite Materials and Their Applicability

Several factors normally enter into the selection of airframe structural materials, including strength, stiffness, density, fracture toughness, fatigue crack resistance, corrosion resistance, temperature limits, producibility, repairability, and cost. However, for supersonic aircraft, operating temperature drives material suitability. Figure 13a indicates how aircraft skin temperature varies with Mach number.[6] Two temperatures are shown—that encountered on the nose, leading edges, and air inlets, and that encountered over the remaining skin surface. The surface temperature for subsonic jets cruising in the stratosphere is just below 0° F[7] but increases rapidly with speed. Additionally, temperatures at the nose, leading edges, and air inlets increase more quickly and reach higher levels than temperatures over the rest of the aircraft surface.

In Fig. 13b, the service temperature limits of the materials addressed in this study are shown along with two metal matrix composites. The epoxies, thermoplastics, and bismaleimides are pretty much limited to aircraft flying in the Mach 2.0 to Mach 2.5 range. Sustained speeds in the Mach 3.0 to Mach 3.5 range will require titanium, steel, or a titanium matrix composite.

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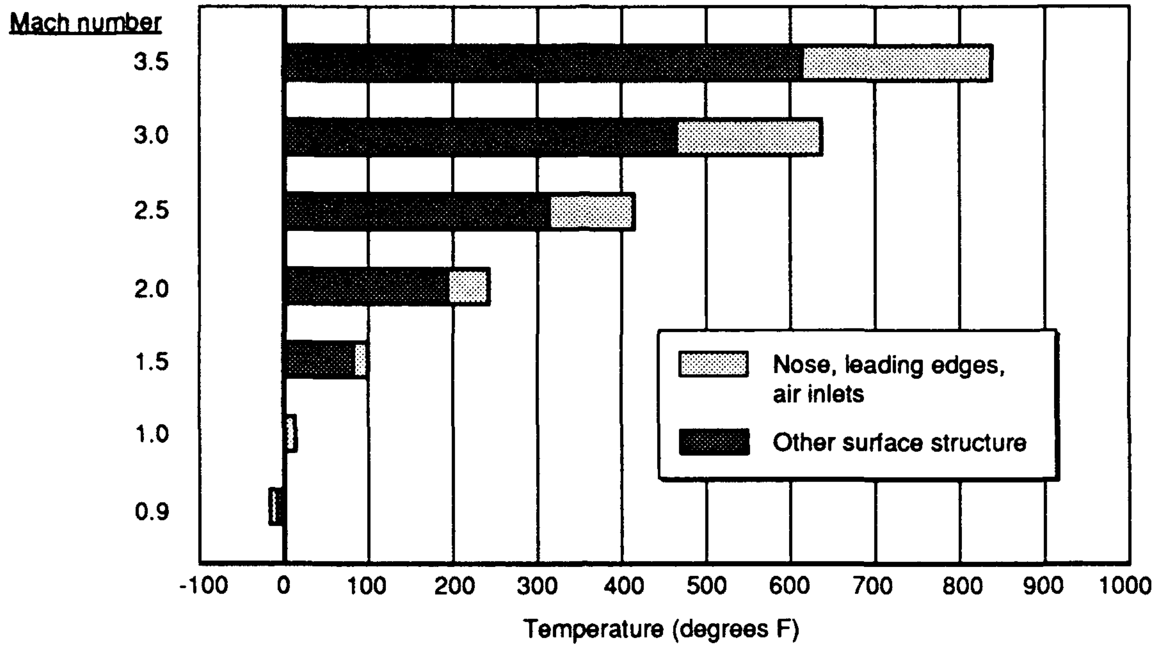
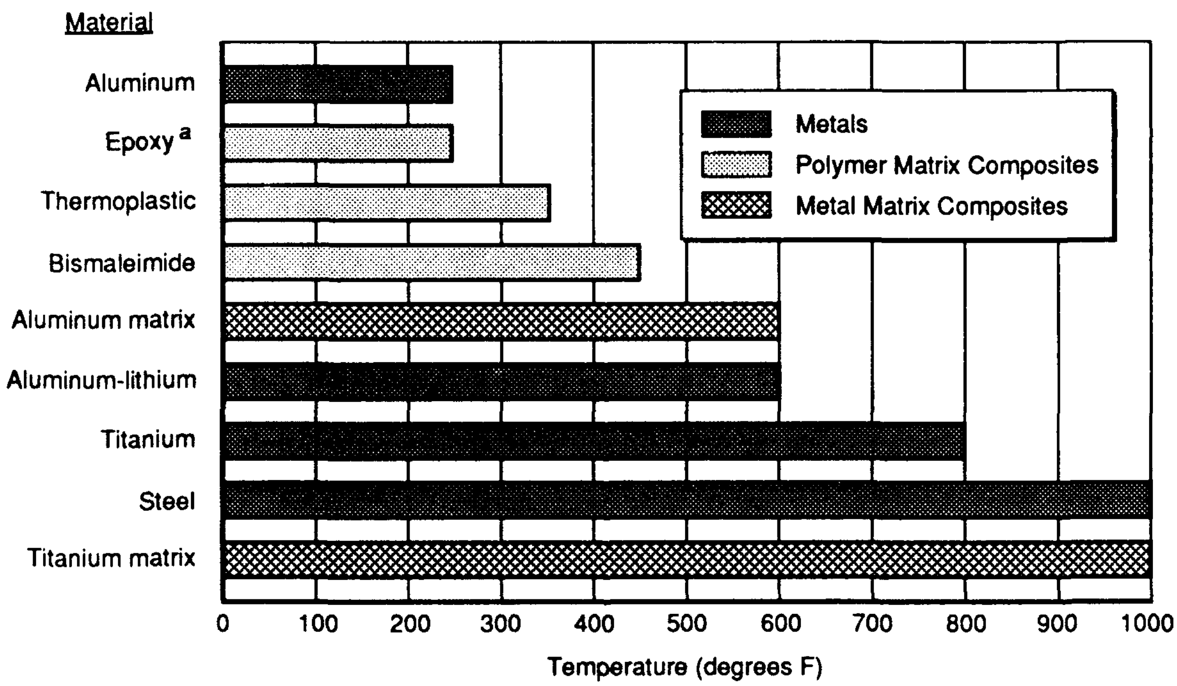
[6]The values shown are, of course, approximations since actual skin temperatures will vary with airflow conditions, surface finish, and atmospheric conditions.

[7]In the case of subsonic aircraft, the skin temperature requirement is determined not by the cruise Mach number at altitude but rather by the ambient temperatures the aircraft encounters.



(A) Aircraft skin temperatures as a function of mach number (cruise at 36,089 feet)



[a] Most epoxies are good to 250 degrees Fahrenheit; a few are good to 350 degrees.

(B) Matrix material service temperature limits (sustained)

Fig. 13—Matching aircraft temperature requirements to material service temperature limits

Composite Material Weight Reduction

Several operational requirements are fueling the incorporation of composites, one of which is a constant desire for increased payload capability. The most prominent advantage composite materials provide is their potential for saving weight. Three characteristics of composites contribute to this. These are:

- The favorable ratios of strength to density and stiffness to density offered by the fiber (see Figs. 10 and 11).
- The elimination of fasteners through a unitized design and cocured parts.
- The efficiency created by aligning the fibers in the direction of load.

The weight saving potential of composites has two aspects: (1) part-for-part weight reduction and (2) resizing, or scaling down of other parts and subsystems.

Table 2 compares the weights of components fabricated from metals and their composite equivalents (part-for-part weight reduction). Experience shows weight savings between 15 and 31 percent have been achieved, with weight savings for primary structure (wings, fuselage sections, stabilizers, etc.) being slightly less than savings for secondary structure (flaps, fairings,

Table 2

SUMMARY OF HISTORICALLY ACHIEVED COMPOSITE WEIGHT SAVINGS

First Flight/ Cert. Date	Component	Weight Savings (%)	Metal Weight (lb)	Metals	Composite Weight (lb)	Composites
1980	AV-8B Wing	19	1,140		924	G/E
1974	B-1 Horizontal Stabilizer	16	3,315		2,780	G/E, B/E, Gl/E, Ti, Al
N/A	B-1 Vertical Stabilizer	18	654		538	G/E
N/A	F-5E Flap	27	34		25	G/E
N/A	F-14 Overwing Fairing	24	66	Al, Ti, Steel	50	G/E, B/E
1968	F-14 Horizontal Stabilizer	18	905	Ti, Al, Steel	825	B/E, Ti, Al, Steel
1975	F-15 Speed Brake	21	112		89	G/E, Al, Ti, Steel
1982	737 Horizontal Stabilizer	22	220		172	G/E
1982	727 Elevator	26	257		189	G/E ribs, spars, skins
1985	DC-10 Vertical Stabilizer	22	1,005		787	N/A
1976	DC-10 Upper Aft Rudder	27	41		30	G/E, Gl/E
1981	L-1011 Aileron	31	111		77	G/E ribs, spars, skins; Kevlar/Nomex trailing-edge wedge
1984	L-1011 Vertical Stabilizer	28	858		622	N/A

SOURCES: Decamp and Johnson, 1982; Zweben, Bader, and Yue, 1987; Dastin, 1986; Warwick, 1987.

elevators, ailerons, fins, and rudders).[8] This was consistent with our industry survey, which indicated weight savings in the 20 to 25 percent range. More advanced composites may yield greater savings. A 1988 OTA study indicated that in the mid-1990s polymer matrix composites may save between 30 and 40 percent over the weight of comparable metal structure, assuming advanced fibers and modified epoxy and thermoplastic matrices.[9]

The discussion above has focused on the weight savings directly achievable by substituting composite structure (which is more structurally efficient) for metal structure. However, even greater savings in aircraft weight are obtainable through a process known as "resizing." That is, because some parts are now made of more structurally efficient composite, the remaining structure has less of a load to support and can be downsized. Moreover, reducing the structure will cause the aircraft to require less fuel to accomplish the same mission. Less fuel means a smaller fuel system is required. Consequently, there is more structure savings because there is less fuel volume to contain. And as the weight comes down, other systems, such as the landing gear and hydraulic system, can also be reduced in size; but the biggest savings comes in the engine(s). Because the aircraft weighs less, it does not require as much thrust to meet any acceleration or specific excess power requirements. Reducing the thrust reduces the weight and volume of the engine(s) and another round of structure and subsystem downsizing takes place.

Whether the weight differences associated with resizing are substantial depends primarily on two factors:

- The aircraft's mission and performance requirements. High performance fighter aircraft usually exhibit much greater sensitivity to resizing than do subsonic aircraft, particularly cargo and bomber aircraft.
- The freedom to make design decisions. The greater the number of design decisions that have been made, the more limited the scope for resizing. By far the most important of these decisions is the engine scale. For example, when the engine scale is fixed and structure weight is removed, the resulting performance gain cannot be "sized out" to save weight (and cost).

Over the long run of aircraft evolution, technology advances, such as the incorporation of composite materials, have generally been used to achieve higher performance, not lower weight. The weight savings discussed above are short-run considerations. A specific aircraft development project will have weight, performance, and cost requirements or goals. Designers will consider many tradeoffs among these various requirements. The process of resizing is used extensively in generating alternative design solutions to meet the requirements, particularly during the conceptual design phase. Because only one design alternative is carried through complete development, it might be more appropriate to identify the lower weight of the final design solution compared to those designs not selected as weight "avoidance" rather than weight "savings." The important point is that the performance capabilities of the final design would have been achieved only at higher weight without the use of the advanced technology.

[8]Some savings presented in the table were based on aluminum structure and others on titanium.

[9]Office of Technology Assessment, 1988, pp. 142–143.

Composite Materials Summary

Tables 3 and 4 have been provided for a quick reference. They summarize characteristics of commonly used matrices (Table 3) and reinforcements (Table 4) for airframe structural elements. Within each category there may be a considerable range in the values beyond those indicated here, depending on the specific matrix and supplier.

Table 3

MATRIX MATERIAL CHARACTERISTICS

	Temperature (° F)		
Matrix	Process	Service Limit	Damage Tolerance
Polymers			
Epoxies[a]	350	250	low
Polyimides	550–650	400–550	low
Thermoplastics	570–800	350	high
Metals			
Aluminum	1000	600	very high
Titanium	1650	1000	very high
Ceramics	1000–3500	3000–4000	low

[a]Some epoxies have service temperature limits of 350° F, but the rule of thumb is 250° F.

Table 4

MATERIAL PROPERTIES (TENSILE)[a]

Fiber/Material	Stiffness MSI[b]	Strength KSI[c]	Density lb/cu in.
Fibers			
Graphite	20–100	300–1,000	.065
Boron	60	500	.07
Silicon carbide	60	500	.07
Glass	10–12	400–600	.07
Aramid (Kevlar)	10–20	500	.045
Metal materials			
Aluminum	10–11	40–80	.10
Titanium	15–18	100–180	.16
Steel	25–30	200–300	.28

[a]Values are in the fiber direction and therefore represent the maximum fiber capability.

[b]MSI is millions of pounds per square inch.

[c]KSI is thousands of pounds per square inch.

METAL ALLOYS

Aluminum-Lithium

Aluminum-lithium (Al-Li) alloys are about 10 percent lighter (and stiffer) and 5 percent stronger than current aluminum alloys. The fatigue properties of Al-Li are also better. Lithium is the lightest metal, and adding small amounts of it to aluminum can greatly enhance mechanical properties. Unfortunately, lithium is highly reactive and can explode during the alloying process. Only recently have these problems been solved sufficiently to produce a commercially available Al-Li alloy. Problems with Al-Li include weldability and batch-to-batch consistency.

Aluminum-lithium might be used as an alternative to composites in some applications (bulkheads, superplastically formed internal fuel tanks) or to save weight if it is substituted for aluminum. According to some sources Al-Li designs might result in 15 percent part weight reductions. In a demonstration program, McDonnell-Douglas replaced conventional aluminum wing skins on the F-15 with Al-Li and reduced weight by 10 percent.

Powder Metallurgy

Powder metallurgical (P/M) processes can offer several benefits with respect to conventional cast or wrought procedures. P/M components can have superior mechanical and microstructural characteristics; they also tend to be more compatible with net shape operations (such as superplastic forming/diffusion bonding and hot isostatic pressing (HIP), which are discussed in the next section) and are more responsive to nondestructive evaluation (NDE) and inspection (NDI) techniques. These processes are being applied to conventional titanium and aluminum alloys (for example the 7000 series), Al-Li alloys, and superalloys such as Rene 95 and IN 100. Alcoa introduced the first commercially available aluminum powder metallurgy alloys in 1980.

Powder metallurgy materials are made, as the name implies, using standard materials such as aluminum and titanium in powder form. Small amounts of other materials are then added, most frequently zinc, copper, magnesium, or cobalt to enhance properties. Because fine particles of the materials are mixed, or alloyed, better properties can be achieved. For example, Al-Li may be made in two ways, using traditional ingot-metallurgical techniques or powder metallurgy. If aluminum-lithium alloys are produced by ingot metallurgy, where the alloying elements are dissolved in molten aluminum and then cast into ingots, the process effectively limits lithium content to 3 percent. Higher lithium contents, and therefore lower alloy densities (which lead to weight savings) may be possible with powder metallurgy. Because P/M materials are alloyed using finer grain structures, a more homogeneous distribution of elements is achieved, making for an optimum combination of strength and toughness. Lithium contents of up to 5 percent instead of 3 percent may be achieved if P/M techniques are used.

Another example is aluminum alloys. An article in *Flight International* stated that compared with ingot metallurgy alloys, aluminum P/M alloys 7090 and 7091 offer higher strength and superior fatigue crack initiation performance. It went on to state that Boeing has used P/M 7090 in its 85 lb landing gear forging for the 757.[10] These materials can be provided in traditional forms. For example, the first commercially available aluminum P/M alloys—7090 and 7091—were identical in form to a cast aluminum ingot, and, according to Alcoa, subse-

[10]Warwick, 1985, pp. 20–22.

quent fabrication steps are essentially alike.[11] They can also be provided in powdered form for use in a near net shape fabrication process. The powder can be loaded into a preshaped form or cast and then it is typically subjected to extremely high temperature and pressure. Because the cast shape is close to that of the finished product, wastage of material is minimized, as is the need for labor. The downside is that mold costs are high.[12]

The primary use for P/M parts has been in jet engine components. These applications have been fairly routine since the early to middle 1970s. More work has and is being performed on applying P/M techniques to other airframe components. Consequently, about six F/A-18 aircraft are gathering operational experience on P/M titanium alloy engine mount supports and arrestor hook supports. The original (metal forging) hook support weighs 260 lb while the hot isostatic pressed P/M part weighs only 55 lb. P/M airframe parts are still somewhat new and in need of further development. They should be considered wherever there is a need for high strength alloys.

[11]Ibid.

[12]Clark and Flemings, 1986, p. 54.

III. DESIGN CONSIDERATIONS AND MANUFACTURING PROCESS TECHNOLOGIES

Composite materials give a designer added flexibility, but this flexibility complicates the design process. The thermal and mechanical properties of composites are different from those of metals. In addition composites are processed differently. Most are not machined, milled, or formed as metals are; shapes are created by placing a series of plies on a form and then permanently setting them under intense heat and pressure for a specified period of time.

COMPOSITE DESIGN AND ANALYSIS CONSIDERATIONS

Composites are two-phase, orthotropic materials. This means that, unlike homogeneous isotropic metals, their mechanical properties can be tailored to meet specific design requirements. For example, it is usually optimal to align the fibers predominantly in the direction of the load. Or it may be desirable to design a structure with a coefficient of thermal expansion of zero in a given direction. Tailoring permits the most efficient use of the material.

Unfortunately, the very two-phase composition that allows structural optimization also introduces the potential for other, less desirable complexities. In large part, these complexities result from the interaction of the two very different components, the fiber and the matrix. The fibers are strong, stiff, and thermally resistant; the matrix is generally much weaker, less stiff, and more susceptible to corrosion and solvents. The fiber-matrix interphase is also complex and can greatly affect the toughness and damage tolerance of the composite. The chemistries and properties of both fiber and matrix must be understood and tradeoffs must be considered to maximize the properties of interest (toughness, stiffness, strength, solvent resistance, etc.).

In addition, much has been made of the fact that material standards have not been established. Material properties can vary from batch to batch and among suppliers. Guidelines relating material properties to final part properties are still being developed. Knowledge is lacking on how structures fabricated with these materials degrade under load. Standards as to the effects voids and cracks per square inch (or in absolute size) have on final part properties or load capability have yet to be fully determined. Designers must consider this uncertainty and potential variability during the design process and additional modelling may be required.

Other important examples of the differences between composites and metals complicate the design process. For example, when a hole is drilled into a flat metal plate, the stress concentration factor at the edge of the hole has a value of three, meaning that the stress at the edge of the hole is three times the "far" field stress (a few hole-diameters away). This is true for any metal and any plate thickness. In a composite plate, the presence of a hole can create stress concentration factors with values anywhere between two and seven, depending upon the laminate configuration. Fiber-dominated layups (those with a high percentage of zero degree plies) tend to have higher stress concentration factors than matrix-dominated layups (those with a high percentage of angle plies). Obviously, such complex responses must be considered in the design of the structure's attachment points.

A delamination is a separation between plies. Under tensile loads, delaminations are not serious, but they can have a tremendous effect on the shear and compressive responses of

composite structures. If a delamination of a critical length were to go undetected, it could propagate under sustained or cyclic loads and ultimately cause the collapse of the structure.

Delaminations often occur during the manufacturing process; a part may be laid up incorrectly, the cure cycle (temperature and pressure) may not be implemented properly, etc. Time and resources must be spent inspecting composite parts for the presence of any delaminations, which, if found, must be assessed in analytical or experimental terms. If any are considered "critical," repairs must be performed and the integrity of the part verified, or the part must be scrapped.

Analytical Models

The analytical tools developed to predict the behavior of composites need further development. Many of these tools are based upon research into homogeneous, isotropic metals and are not generally applicable to composites.

The analytical tools that have been developed to predict the damage growth behavior (and residual part strength) of composites are still in early development. They are hampered by the fact that there are no universally accepted theories relating to the damage growth and failure of composite materials. While crack growth in metals is fairly well understood, the rules of delamination growth in composites still elude both industry and academia. Thus, any analytical results must be heavily supplemented with rigorous (and often expensive) testing programs that apply to the specific part design and loads of interest.

Design

A cautious and conservative approach has been used in introducing composites into critical structures. Because of the analytical inadequacies, wide scatter in batch-to-batch prepreg properties, and the variability between manufacturing processes, designers have imposed conservative design strain values on the materials. Stress analysts impose high "factors of safety" to the stresses they predict for the part. These conservative practices add weight to the part. With greater expertise, development of new analytical tools, and production of tougher materials, confidence will increase and these weight penalties will be reduced.

The premise behind unitized design philosophy is that the most benefits (structural weight and cost savings, performance, range, payload, etc.) will accrue when a weapon system is initially designed with composites in mind.[1] In this way, there is maximum compatibility among all parts, and maximum use of automated manufacturing processes is possible.

A basic tenet of this philosophy is that composite designs will rarely be cost competitive with metals unless the part and fastener count (and therefore tool count) can be drastically reduced. Unitized designs will tend to emphasize such techniques as cocuring and integral stiffening.

As designs become more unitized, tradeoffs begin to appear. On the one hand, repair and inspection procedures become more complicated; certain areas of the system will be more difficult to access, larger pieces will have to be handled and inspected, the expense of a mishandled autoclave run increases, etc. On the other hand, certain elements of risk have been reduced; if fewer holes are drilled for fasteners, fewer cracks will be initiated and cyclically grown. There will also be fewer tolerances accounted for.

[1]A unitized design refers to the philosophy of designing and building a structure in one piece, thereby eliminating the need for fasteners. Traditionally these structures were manufactured in separate pieces and then mechanically aligned, shimmed, and fastened. The unitization of the piece then can save much of the alignment and assembly work and makes the piece less likely to contain undetected flaws associated with drilling holes, etc.

It is difficult to assess where the limit of unitized designs will occur. One limit may simply be autoclave size; 8 ft (diameter) by 20 ft (length) autoclaves are common (although larger ones exist). Another unknown is the effect of thermoplastics. If they live up to their potential, repairs should be much easier (and quicker) to perform in the field than for their thermoset counterparts; this trend would encourage the production of larger, more integral parts.

COMPOSITE FABRICATION TECHNIQUES

Fabrication costs are driven by the design requirements of the structure. These requirements determine a set of candidate materials that determine a set of fabrication processes. Generally speaking, superior performance requires more costly materials and fabrication techniques. Primary structures, such as wing or stabilizer skins, are highly loaded, flight-critical structures that must be resistant to fatigue and environmental effects. Secondary structures do not carry critical loads, and the consequences of failure are less dramatic. Therefore, primary structures demand high performance materials and are expensive to make while secondary and more moderately loaded parts may be able to take advantage of lesser materials and cheaper, quicker manufacturing processes.

Table 5 provides an overview of composite manufacturing process temperature and pressure control requirements. Also included are rough estimates of tooling, production, and material costs. Each process is applicable to a limited number of materials, and the requirements of each material vary in terms of temperature and pressure controls.

Part shape and size put additional constraints on the applicable fabrication technique employed. Table 6 presents information on the suitability of certain manufacturing techniques to various component forms. For example, medium to large planar panels such as wing and stabilizer skins, fuselage skins, and doors, because of their form can be fabricated using autoclave curing, elastic reservoir molding, thermoforming (thermoplastics), hot stamping, or rapid cure (thermosets).

Primary structures meet strenuous stiffness and strength needs by controlling the orientation of the fibers with respect to the design loads. This requirement for carefully controlled fiber alignment is perhaps the most important cost driver with respect to composite parts, has slowed the adoption of automation, and has forced the use of expensive manual labor for many critical parts.

Automated fabrication procedures have been extensively studied and developed in the last ten years. Some contractors have been, or are, in the process of installing expensive systems that they hope will beneficially affect their production rates and costs. These systems are starting to come on line now, but it is difficult to predict what problems will arise and how truly cost-effective such systems will be.

"Implementing automation in this [airframe] industry has not been easy given the limited production of aircraft, the high cost of automation equipment, and the complexity of typical aircraft composite parts."[2] Complex operations requiring great dexterity will probably remain manual procedures. However, some operations, such as ply cutting and layup (for certain parts) will very likely be automated in the near future and the cost/time savings should be substantial.

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[2]Vaccari, 1987, p. 88.

Table 5

PROCESS MANUFACTURING REQUIREMENTS AND COSTS

Process	Materials	Close Pressure Control	Close Temperature Control	Post Cure	Tooling Costs	Production Cost	Material Cost
Autoclave curing	Glass, Kevlar, graphite fabric; thermosets, thermoplastics	Yes	Yes	May be required with thermosets	High	High	Low to high; depends upon fiber/resin choices
Elastic reservoir molding (ERM)	Glass, Kevlar, graphite fabric; foams, epoxy resins	Yes	Yes	May be required	Low	Low	"
Thermoforming thermoplastics	Glass, Kevlar, graphite fabric; thermoplastics	Yes	Yes	No	Low	Low	"
Injection molding	Glass, graphite, chopped fibers; thermoplastics, thermosets	Yes	Yes	No	Depends upon part complexity	Low	"
Hot stamping	Glass, graphite, Kevlar fibers; thermoplastics, thermosets	Yes	Yes	No	Moderate	Low	"
Rapid cure thermosets	Glass, graphite, Kevlar fibers; thermosets	Yes	Yes	No			
Pultrusion	Glass, graphite, Kevlar fibers; thermoplastics	No	Yes	No	Low	Low	"
Filament winding	Glass, graphite, Kevlar fibers; historically with thermosets; thermoplastics developing	No	No	Some applications	Low	Low	"

SOURCE: Mahon, personal communication, July 1989.

Table 6

SUITABILITY OF MANUFACTURING PROCESSES TO ALTERNATIVE MANUFACTURING FORMS
(Examples listed are intended to be illustrative)

	Form of Manufactured Component					
Process	Large Integral Structure[a]	Highly Contoured Parts[b]	Med/Large Plain Panels[c]	Closed Sections[d]	Open Sections[e]	Detailed Parts[f]
Autoclave curing	Yes	Yes	Yes	Yes	Yes	Yes
Elastic reservoir molding (ERM)	No	No	Yes	No	Yes	Possible
Thermo-forming thermo-plastics	No	No	Yes	No	Yes	No
Injection molding	No	No	No	Yes	Yes	Yes
Hot stamping	No	No	Yes	No	Yes	Simple Brackets
Rapid cure thermosets	No	No	Yes	Yes	Yes	Simple Brackets
Pultrusion	No	No	No	No	Yes	No
Filament winding	Yes	Yes	No	Yes	No	No

SOURCE: Mahon, personal communication, July 1989.
[a]Includes fuselage skins with stiffeners, wing skins with stiffeners, and bulkheads.
[b]Includes leading edges and fairings.
[c]Includes wing skins, stabilizer skins, fuselage skins, and doors.
[d]Includes closed hat section stiffeners, ducts, and piping.
[e]Includes stiffeners ("L" shaped and "Z" shaped), beams, ribs, and frames.
[f]Includes fittings and brackets.

The Conventional Fabrication Process for Laminates

A simple and generic description of the fabrication steps for composite parts follows:

- Tool and material preparation.
- Pattern (ply) cutting.
- Material transfer and orientation (layup).
- Debulk.
- Precure and vacuum bag preparations.
- Cure and postcure.
- Assembly.
- Inspection.

Tool and Material Preparation. Most tools used to produce composite parts are metallic. Long-term production tools are usually made of steel or nickel, but aluminum and occasionally plastic tools have been successful as well. Plastic, composite, rubber, or aluminum tools are often not durable enough to survive long-term production runs and mishandling. However, there are circumstances where composite tools may be necessary.

Tooling costs for composite materials can be high partly because of the poor machinability characteristics of composites. A metal part, by comparison, does not require as strict tool tolerances because machining away excess material is easier. Duplication of expensive tools because of long autoclave cure cycles is often necessary. Any thermoplastic parts that must be processed in an autoclave will incur high tooling costs because of the high temperatures (600°F) the tools and the autoclave must endure.

Another tooling cost driver is the mismatch between the thermal expansion of the composite part and that of the metal tool during the cure cycle. Steel, and sometimes aluminum, tools are often necessary because of strict processing temperature and tool durability requirements. However, metal tools will tend to expand more than the composite part. As long as processing temperature and part tolerance requirements are not too high, the warping effect may not be critical and acceptable parts may be produced from metal tools.

The trend in composites is indeed toward materials with higher processing (and service) temperatures. Bismaleimides, polyimides, and thermoplastics all have higher manufacturing temperature requirements than conventional epoxies. Therefore, the thermal mismatch between composite parts and metal tools becomes more important; excessive mismatch can damage both the tool and the part. Thus, different (and probably more expensive) materials must be used for tool fabrication. However, to some extent, this problem can be alleviated by more sophisticated tool designs and analytical models. These approaches can help minimize the deformation mismatch before tool production.

Another alternative for parts that require high processing temperatures is tools fabricated from composites. This concept has the advantage of a close thermal match between the tool and the part, which may be necessary for complex shaped structures. During the cure cycle, such tools expand and contract in better harmony with the part since they are made of the same material.[3] Unfortunately, composite tools suffer from a number of deficiencies: Their durability is in question, they are slow to heat up, and they are very expensive (one contractor estimated some of his composite tools to be 20 times more expensive than metal counterparts). One reason for this expense is that even more tools are required to make the composite tools.

Thermoplastic materials will further tax conventional metallic tool design; thermal mismatch effects would be even greater because of the higher processing temperatures required for thermoplastics. Conventional aluminum tools will probably not be possible since temperatures will exceed 600°F. The durability of steel tools would also be affected by the substantially higher processing temperatures. Thus, research into composite tools is a necessary and expensive effort.

Long layup times can also be a tooling problem. Ideally, expensive production tools should be "cooking" nearly all the time; but delays can accrue if a large, complex part requires perhaps a week to lay up. To minimize the effects of long layup times, some companies have several cheap plastic male tools for each female production tool. Several parts can be laid up on the male tools simultaneously. Then these parts can be easily transferred, one at a time, to the female tool for curing. In that way, the cumulative production time (over several units), or the number of expensive autoclave-capable tools, can be greatly reduced.

[3]Another advantage to block graphite tools is that they can be machined using numerically controlled machine tools just as a conventional machined metal part is formed.

Pattern Cutting. *Manual* pattern (ply) cutting is a slow process, about 100 inches per minute for a trained composite technician. This technique is most often used for small and medium sized parts with complex contours. Such parts are not yet amenable to automation. Manual pattern cutting makes it difficult to cut multiple plies with accuracy, the process is slow and labor-intensive, and costly inspection procedures must be implemented. Many contractors have already purchased and are using automated pattern cutting facilities wherever possible.

Automated pattern cutting procedures include Gerber knife machines, waterjets, lasers, and chisel cutters. Cutting speeds are much higher than for manual labor: 300 inches/minute and multiple ply capability (four plies of graphite/epoxy) for Gerber machines; much higher limits are planned for the chisel cutters. All of these procedures are numerically controlled and highly accurate even for large plies, which are difficult to handle using manual techniques. Automated ply cutting should eliminate some of the costly inspections required for manual procedures. These systems should be able to handle thermoplastics and thermosets in exactly the same way. Automated ply cutting facilities are already common among the major composite part manufacturers.

Material Transfer and Orientation (Layup). The layup process, by one estimate, represents 30 percent of all composite labor costs. As mentioned earlier, much of this is driven by requirements for fiber alignment. It is in the layup process that automation may have its most influence with respect to cost savings.

Hand layup is estimated to account for over 90 percent of all parts in the aircraft industry. Hand or manual layup is often necessary because complex shapes are not amenable to automated tape layers. This means that many parts have contours that exceed the natural compliance of the prepreg tape. Hand layup procedures can be lengthy, error-prone (despite the attention of the technician), and costly. For example, the AV-8B wing skin, which is composed of about 200 individual plies spanning an area of nearly 300 square feet (10 × 30), requires about 160 labor-hours of hand layup. Research estimates that humans are likely to make mistakes in the layup process if there are more than 20 separate plies to form. Thus, time and money must be spent monitoring hand layup parts and correcting (if possible) any mistakes.

Automated layup can often be performed by tape laying machines for larger, mildly contoured parts such as wing and stabilizer skins. For such parts, automated tape laying machines can operate many times faster than manual procedures. Another advantage is that plies laid up by machines often do not need separate debulking procedures because of the high compaction capabilities of the machines. Many parts have contours and ply designs that are too complex to be readily handled by these machines. However, there are new programming tools that will enhance compatibility between the parts and the tape laying machines.

Many new designs are benefiting from "natural path" programming. These software routines graphically display the natural paths that the plies will want to follow given the part geometry, the type, width, and thickness of the prepreg material. Designs can then be evaluated and, if necessary, changed, to more fully comply with the use of automated tape layers.

Thermoplastics (and advanced thermosets such as some bismaleimides) have poorer drapability and tack than epoxies. Therefore, some additional heat during layup is required to enhance formability. In manual layup procedures, bismaleimides can be softened with heat guns (90–130°F). However, thermoplastics require as much as 650°F for forming. Automated tape laying machines for these materials use warm (bismaleimides) and hot (thermoplastics)

dispensing heads to lay up the plies. Such machines are new for thermoplastics and currently confined to producing flat laminates. This limitation is somewhat mitigated by the fact that a flat thermoplastic laminate (produced by an automated tape layer) can be reheated and reconsolidated against a mold of the required curvature more swiftly and accurately than if the entire layup process were done by hand. State of the art, numerically controlled tape laying machines are estimated to cost between $2,000,000 and $3,000,000. A hot head tape layer, required for thermoplastics, would be at the higher end of the range, perhaps more.

Ply Management is an issue of growing importance as automation and production rates increase. Thousands of plies will have to be managed on a daily basis. Industry is investing in automated, computer-managed systems that must cope with many issues: coding of plies as to material type and fiber orientation, which part the plies belong to, sequence in the layup, elapsed time from removal from the freezer (thermosets), kitting (bagging plies for freezer storage), ply retrieval, and preparation of layup books.

The Automated Ply Laminating System (APLS), being developed at McDonnell Douglas in St. Louis, will produce flat ply collations, which can then be manually formed about the curing tool. Plies will be cut, labeled, kitted, stored, retrieved, and collated with minimal human assistance.

Debulking. Debulk operations may be required before cure to compact the plies and to eliminate any interply gaps or voids. Thick laminates may require on the order of 10 debulk cycles to properly seat the plies onto the tool. This time-consuming process can sometimes be eliminated or reduced by the use of automated tape laying procedures that exert high pressures on the uncured part.

Precure and Vacuum Bag Preparations. Matrix bleeder materials are required to ensure that the proper amount of excess resin is bled out of the laminate during the cure cycle. This is important because excess resin will only degrade the final part properties as well as add extra weight. Such materials as peel ply, porous *teflon*, nonporous teflon, and fiberglass bleeder plies are placed in intimate contact with the uncured laminate after it has been laid up on the tool. The laminate assembly is then ready to be sealed in a vacuum bag.

Bagging procedures are extremely important for autoclave cures. According to the *DOD/NASA Structural Composites Fabrication Guide*, improper bagging and sealing operations are probably the most prevalent cause of scrapped parts. If the vacuum seal is lost during the cure cycle, the pressure pushing the laminate against the forming tool vanishes. Without the support of the tool, the plies will shift, and part tolerances (and contours) will not be met. Also, any volatile gases generated during cure will not be drawn out of the laminate, which increases the possibility of undesirable voids and delaminations in the cured part. Therefore, careful bagging and sealing procedures must be maintained and checked.

Cure/Postcure/Consolidation. A cure cycle is a prescribed schedule of temperature and pressure that is required to process the thermosetting matrix material of the laminate to its final hardened state. This cycle may be specified by the material vendor, or the contractor may use a cycle that has been developed in-house.

A typical cure cycle for graphite/epoxy lasts about 3–4 hours, reaches pressures of 50–100 psi and temperatures of 350°F, and requires a postcure of several hours. Graphite/polyimide laminates require a maximum pressure of about 200 psi, a maximum temperature around 650°F, and a postcure of nearly 16 hours. After curing, the laminate is removed from the tool and stripped of the matrix bleeder materials. Thermoset parts are often postcured (temperature only) in ovens. The purpose of the postcure is to achieve maximum crosslinking and strength in the thermosetting matrix.

Past programs simply recorded the temperature and pressure levels as a function of time, within (for example) the autoclave to show compliance with the recommended cure cycle. Future emphasis will be placed on monitoring the key physical and chemical parameters of the actual matrix material. This should help to eliminate some of the wide scatter in the mechanical properties of composites since it is the state of the matrix that needs to be properly monitored, not the pressure and temperature levels in the autoclave. Ultimately, if the scatter is reduced, confidence will be increased and design penalties for composites reduced.

Thermoplastic materials are said to consolidate rather than to cure. This process is much quicker (minutes), but temperatures can reach 800°F, depending upon the material. Also, thermoplastics do not require a postcure cycle. These materials will require more capable and durable manufacturing equipment (autoclaves, ovens, presses, etc.) than those currently used for epoxies.

Autoclave curing is the most common, state-of-the-art method used for producing high quality composite parts with thermosetting resin matrices. Autoclaves are expensive; a single autoclave 20 feet (diameter) by 50 feet (length) can cost $7,000,000 (for use with thermosets) or $11,000,000 (for use with thermoplastics). (As autoclaves get larger, they cost more.)

Assembly. *Mechanical assembly* procedures include drilling holes, trimming, cutting, sanding, bonding, cocuring (practically speaking, this is part of the cure cycle), fitup, mechanically fastening, etc.

Some of these operations are more difficult (and more costly) with composites because of their brittle nature and historically low matrix toughness (of thermosets). This makes composite parts very sensitive to low-energy impacts (such as the accidental dropping of a tool). Although there may be no visible damage on the surface, delamination and matrix cracking may exist in the interior of the laminate. Unseen internal damage can have an enormous negative effect on compressive strengths. Composites are also sensitive to the drilling of high tolerance holes; matrix cracks and fiber damage can be induced around the edges of holes. Current research emphasizes the toughening of existing thermosetting matrices, the exploitation of the inherently tougher thermoplastics, and the development of appropriate thermoset/thermoplastic hybrids. These efforts should yield beneficial results with respect to decreased levels of inspection, increased tolerance to low-energy impacts, and reduced maintenance requirements.

Assembly can be a considerable portion of the total part cost; estimates range as high as 40 percent. One reference concluded that drilling and countersinking holes between graphite/epoxy parts can be four times (per hole) as expensive as between aluminum parts. Fastener costs and installation can be between one and five times as expensive as for aluminum parts. The expense of joining graphite/epoxy to aluminum structures is largely driven by the galvanic reaction that occurs when aluminum and graphite are in intimate contact. Corrosion is the consequence of any such contact, and thus more expensive titanium fasteners are used instead of aluminum. Other concepts for avoiding this galvanic link include the layup of a single fiberglass ply, which acts as insulator between the incompatible materials.

After all of the machining is done, the separate parts are assembled in the fitup procedure. Fitup is a good test of the fabrication system since it must account for and integrate all of the accumulated tolerances, mistakes, and defects. Fitup and assembly costs can be as much as 15 percent of total manufacturing cost.

Cocuring is an operation in which multiple parts may be both cured and adhesively bonded to one another during a single, often autoclave, run.

Composites compete more effectively, in an economic sense, with metals if designers can reduce the number of parts in an assembly. This "integral structure" approach reduces not only the total part count but also the number of holes that must be drilled and the number of fasteners that must be installed, expensive operations for composites. Generally weight is also reduced.

Detailed parts can be cocured, but major assemblies are mechanically fastened. Many subassemblies that are currently secondarily bonded will instead be cocured by the mid-1990s. Examples of cocuring indicate that manufacturing costs can be reduced by as much as 40 percent.

Secondary structures, where the structural consequences of failure are fairly low, are reasonable candidates for cocuring. Heavily loaded primary structures will probably not be cocured because of inspection requirements. In primary structures, it is usual and prudent to nondestructively inspect all parts in detail before assembly. This is not possible with current cocuring techniques. Thus, primary structures are likely to use secondary bonding operations or mechanical fastening methods. A recent example of cocuring is the AV-8B fuel tanks.

The trend toward cocuring is motivated by the desire to reduce the number of fasteners and assembly operations. However, some do see risks associated with this strategy: "Large multi-component assemblies will increase the tool size and the number of details. Bonding fixtures with 1,000 separate details will generate an accountability, storage, and replacement problem."[4] More integral structures will affect supportability as well. Reliable, on-aircraft repair techniques will be required since these larger, unitized parts will not be as easy to remove, repair, or replace.

Advantages of Cocuring

- Lower part count.
- Smaller inventory.
- Fewer holes and fasteners.
- Lower manufacturing costs.

Disadvantages of Cocuring

- Reduced access for inspection.
- Increased complexity for tool design and integration.
- Increased risk if autoclave performs improperly.
- Increased expectations on field-level, on-aircraft repair skills.

Inspection. According to the *DOD/NASA Fabrication Guide* (Meade, 1982), "Many defect types can affect the quality of a composite structure and no single nondestructive test can find and isolate all of them." Common defects include delaminations, foreign matter (inclusions), high porosity (too little matrix material), honeycomb core damage, moisture, fiber breaks, and matrix cracks. These flaws may originate during the manufacturing of the part, as a result of accidental mishandling during normal airframe use and maintenance, or as a consequence of battle damage.

There are many methods of nondestructive inspection, but each has its limitations. Table 7 describes the sensitivity of several NDI methods to different flaw types. An entry of "VG" indicates good sensitivity and reliability between the particular flaw type and NDI method. An entry of "G" indicates less reliability or limited applicability, and an entry of "L" indicates

[4]Spinks, 1986, p. 425.

Table 7

SENSITIVITY OF NDI METHODS TO DIFFERENT FLAW TYPES

	NDI Method								
Flaw Type	Ultrasonic Transmission[a]	X-ray Radiography	Neutron Radiography	Laser	Thermal Infrared	Tap Test	Acoustic[b]	Eddy Current	Visual
Porosity	VG/G	VG	L	—	—	—	G	—	—
Foreign material	VG/G	G	L	—	L	—	—	—	—
Delamination	VG	—	—	—	G	—	L	—	—
Matrix cracks	L	G	—	—	—	—	G/L	—	—
Fiber breaks	—	VG	—	—	—	—	G	G	-
Impact damage	G/L	—	—	G	G	G	L	—	L
Skin/skin disbond	VG/G	G	G	VG	VG	VG	—	—	—
Skin/core disbond	VG/G	L	L	VG	VG	G	G	—	—
Core damage	VG	VG	—	G	G	—	—	—	—
Water intrusion	L	G	VG	L	L	—	G	—	—

SOURCE: Meade, 1988.

[a]Ultrasonic transmission includes four types of tests: through transmission, pulse echo, angle, and resonance.

[b]Acoustic includes two types of tests: emission and ultrasonic.

— not applicable.

VG: Good sensitivity and reliability; good candidate for primary method.

G: Less reliability or limited applicability; may be good backup method.

L: Limited applicability; may provide some useful information.

limited applicability. Every flaw must be well characterized so that engineering decisions with respect to repair or part replacement can be made. The inspection of composite structures is complicated in part because composites have so many different flaw types (compared with metals, which are mainly inspected only for cracks or corrosion). Several of these flaw categories are identified in Table 7. Naturally the seriousness of the flaw depends on its size; however, the more serious flaw categories are delaminations, fiber breaks, porosity, and crushed core (included under core damage). Additionally, each inspection technique (also shown in Table 7) is not sensitive, or has limited sensitivity, to each category of flaw. Thus, several techniques must be used on individual parts depending on their specific design and criticality. Levels of inspection are determined by such factors as part type (primary, secondary, nonstructural); stress analysis; part complexity; and part history. As the materials, manufacturing processes, and analytical tools develop, the number of costly inspection procedures should be reduced.

Ultrasonic transmission methods are the most commonly used method with epoxy laminates. A quick scan of Table 7 shows that nine out of the ten flaw types listed are reasonably sensitive to at least one of the ultrasonic procedures. Currently, there is an evolving interest in thermal IR procedures displaying varying part quality in terms of colors. Magnetic resonance imaging (MRI), which is finding increasing use in medical fields, is also likely to be applied to inspection of aircraft parts. Many contractors now have automated inspection stations.

Specialized Fabrication Processes

The traditional and most common method of composite fabrication in the aircraft industry encompasses manual layup in combination with autoclave cure. However, there are specialized layup, cure processes (thermosets), and consolidation techniques (thermoplastics) that are used when possible. Below we describe the following techniques:

- Filament winding (layup).
- Braiding (layup).
- Vacuum bag curing.
- Thermal expansion curing.
- Hot roll forming.
- Injection molding.
- Compression molding.
- Hot stamping.
- Thermoforming.
- Pultrusion.
- Hydroforming.
- Elastic reservoir molding.

Filament winding has primarily been used for large, axisymmetric structures such as solid rocket motor cases, pressure vessels, and helicopter blades. However, other shapes, such as beams and channel elements, are now being considered; filament wound fuselages might also become practical.

During filament winding, individual fibers or narrow prepreg tapes are dispensed from a translating head to a rotating mandrel. Fibers are pulled through a resin bath just before contact with the tool. Angles of the fiber relative to the rotation axis of the tool are controlled. Both the density and mechanical properties of the part can be affected by a predetermined tension in the fibers or tape. This process has been performed with thermoplastic materials, but the high temperature processing requirements pose some manufacturing problems. Parts formed in this manner can either be autoclave cured or room temperature cured.

Braiding also has been used to produce long and continuous lengths of simple (can be tapered) cross sections. Preforms of the reinforcing fibers are produced and then impregnated with the matrix material. Multidimensional braiding through the thickness direction confers greater damage tolerance characteristics to the part. This technology is still labor-intensive.

Vacuum bag curing is very similar to autoclave curing. The maximum external pressure is limited to 14.7 psi (atmospheric). Heat is usually supplied by an oven. Vacuum bag curing produces parts of lesser density (and cost) than those cured in autoclaves.

Thermal expansion curing is another fairly cheap method of creating the necessary curing pressures by operating on the principle of differential thermal expansion. Uncured plies are wrapped around rubber molds and then placed inside a metal cavity. As the oven temperature increases, the rubber will try to expand more than the metal cavity. This restrained thermal growth creates a pressure that is transferred from the rubber to the metal cavity through the laminate. This process is cheap, but care must be taken in the design of the fixtures so that the proper pressures are created.

Hot rolling of thermoplastics is related to roll forming in metals. Roll forming runs flat metal sheet through a series of rolling dies that gradually shape the metal into the desired cross-section. This process is intended for the production of long and continuous members of

constant cross-sectional shape. Thermoplastic components can be produced by heating the dies sufficiently to soften the resin and form the part.

Injection molding produces parts by injecting a measured quantity of resin and chopped fiber into a molding die cavity that defines the shape of the part. Consolidation occurs under the application of heat and pressure. This process is adaptable to both thermosets and thermoplastics. Molds are expensive, so parts that are required in fairly high numbers would take the greatest advantage of this procedure. Current parts can be complex but generally small. Research with respect to larger parts is ongoing. One problem is the prediction and control of short fiber orientations; lack of control can lead to unequal distributions of mechanical properties. Current cycle times are generally less than five minutes.

Compression molding has been used with short fibers and thermosetting matrices. It is being extended to include thermoplastics. Parts are usually simpler in shape and larger in size than those produced by injection molding. The raw materials are placed in a heated mold cavity and then squeezed in a hydraulic press. Cycle times are in hours, as opposed to injection molding cycle times of minutes.

Hot stamping is related to stamping of parts in metal sheets. To adapt this process to thermoplastic materials, heat (up to 700°F for PEEK) must be supplied. Cycle times are generally less than five minutes. Stamping pressures might reach 5,000 psi for some parts. This process is suitable for chopped fiber reinforcements.

Thermoforming is related to hot stamping but is used when the fibers are required to maintain specific orientations. Again, the basic process is that the laminate is preheated and transferred to a mold in a press where the final part shape is determined. Typical pressures range from 15–100 psi.

Pultrusion is useful for producing lengths of members with constant cross-sections. It uses guides, shape dies, and heat to produce long parts with constant cross-sections, which are then automatically cut to desired lengths. The process consists of pulling dry fibers through a resin bath and then through steel dies that define the shape of the part and control the amount of resin in it. Heat must be supplied to cure the part. This process is economical but is currently limited to straight pieces. With more development, components with varying cross-sections should be achievable. Thermoset parts can be reasonably produced using pultrusion, and it is readily adaptable to thermoplastic materials. However, experience with pultrusion for thermoplastics is still limited.

Hydroforming fiber-reinforced thermoplastics is also based upon experience with metals. The procedure uses a hydraulic press fitted with a flexible pressurizing medium on the upper platen and a rigid form block on the lower platen. The composite part is preheated to the proper forming temperature (perhaps in an oven) and is transferred to the form block. The press is closed and the laminate is formed into the desired shape. While the pressure is applied, the resin cools below the forming temperature and thus will retain the proper shape when removed. Press time should be on the order of half an hour or less.

Elastic reservoir molding (ERM) produces sandwich components consisting of a rigid, polyurethane, foam core with reinforcing facesheets. Graphite, Kevlar, and glass fibers have often been used in combination. The foam is preimpregnated with a known amount of resin, which is transferred to the facesheets under pressure. The resins used are thermoset epoxies or polyesters. This process can use existing hydraulic presses. Rigid molding dies are attached to the upper and lower platens of the press. The foam core (including resin) and the reinforcing facesheets are placed between the dies. The molding dies are closed by platen pressure and the resin is squeezed through the fabric facesheets. Excess resin and gases are ventilated from

the dies. The application of molding pressure and heat cures the part. ERM can also be used with prepreg tapes or fabrics.

Automation of the Manufacturing Process

This section discusses where, and to what extent, factory automation will penetrate composite part fabrication in the airframe industry. As little as ten years ago, all aspects of the production of composite parts were dominated by manual procedures. Since that time automation has made, and should continue to make, great progress in some areas of the manufacturing process (e.g., material cutting, ply management, and nondestructive testing). However, progress is likely to remain limited in other production activities, particularly those confined to operations on small or complex parts (as is often the case in layup procedures). Advances in automation seem to be largely driven by individual computer systems so that information can be quickly and accurately disseminated. Our research found that most manufacturers believe automated procedures will play an ever increasing role in their ability to competitively produce advanced systems. However, there is a range of opinions and expectations as to the extent these automated procedures will prevail in the 1995 timeframe.

Using the cost information reported in Tables 14 through 18, the cost breakdown for producing a composite part is approximately 10 percent for engineering, 10 percent for tooling, 60–65 percent for manufacturing, 10 percent for quality assurance, and 10 percent for material costs. Although the other cost elements are important, manufacturing offers the most leverage for automation (assuming automation is equally feasible across cost elements). Consequently, much of this discussion (and the discussion in the open literature) centers on automating manufacturing

The steps in the fabrication process, as listed in the previous section, are cutting, material transfer and orientation (layup), debulk, precure and vacuum bag operations, cure, and postcure. Material transfer and orientation by far is the dominant contributor to fabrication cost. This can be seen in Table 8, which lists the distribution of the fabrication effort for particular components.

Automated Layup. The accuracy of fiber orientation, which may change between plies or series of plies, is crucial to part performance.[5] Small, highly contoured parts may always

Table 8

BREAKDOWN OF FABRICATION EFFORT
(Percent of fabrication hours)

	Fighter Stabilizer Skin[a]	Wing Skin[b]
Cutting	17	3
Material transfer and orientation (layup)	40	77
Debulk	17	7
Precure and vacuum bag operations	11	3
Cure and postcure	14	10

[a]Meade, 1982.
[b]Huttrop, 1985.

[5]Vaccari, 1987, p. 88.

require the dexterity of the human hand. Compound contours also add complexity, and automated tape layers are restricted to large, slightly contoured components using unidirectional material. Automated layup of, for example, wing skins is generally limited to unidirectional tape widths of less than a foot (often only three inches); fabrics are not currently used in this process. Also, the automated layup procedure for such a part is really combining the separate steps of ply cutting and layup; the machine, in essence, makes its own plies as it moves about the tool, automatically unwinding and cutting the roll of tape according to its preprogrammed instructions.

According to a McDonnell Douglas study, automated tape layers are precluded from application to sections smaller than 4 by 4 feet.[6] Several studies have found that setup time and diseconomies with having to cut the tape and change direction are major reasons why tape layers are not cost-effective on both small parts and parts with irregular shapes that require the tape layer to lay short courses. "The principal factor limiting more widespread use of tape layers, however, is that most composite components have greater contour than the compliance of the tape, which ideally should butt edge-to-edge across a mold surface."[7] For whatever reason, we observed that automated tape layers were in limited use during our plant tours. Our observations are confirmed by an article in *American Machinist and Automated Manufacturing*, which stated, "Despite the availability of tape layers, however, manual layup prevails in the aircraft industry. . . . About 95 percent of all layup jobs . . . are performed by workers by hand or using rollers and other aids, mainly because the complex contours of most parts are not amenable to tape layers. And it's a long and tedious task."[8]

Future applications in this area may include the use of machine vision systems. They will be able to determine if plies have been oriented correctly and if any unwanted foreign matter has attached itself to the plies.

For smaller parts (less than 4 x 4 feet), or those with compound curvatures, flat ply collation combined with hand layup may be an alternative to automated tape laying.[9] And in fact, McDonnell Douglas Aircraft found that on their AV-8B program, flat ply collation combined with hand layup was more cost-effective than an automated tape layer. With flat ply collation either tape or fabric may be used. The limitation on flat ply collation is a radius of curvature no smaller than 10 inches.

Pultrusion and filament winding are other promising methods of automating the layup process. Both processes "create" their own plies by directly orienting and placing individual fibers. Pultrusion is limited to those components with constant cross-section, and filament winding is limited to axisymmetric sections. Thermoplastics may be more promising for pultruding than thermosets because once pultruded the thermoplastic piece may be heated and reformed into more complex shapes by twisting, bending, or flanging the component.[10] Filament winding, while cheaper than hand layup, is not cost-effective for smaller quantities where setup costs becomes a driver.[11] If layup is automated, debulking is not necessary since pressure is applied during the layup process.

Automated Ply Cutting and Ply Management Systems. The cutting of prepreg fabrics and tapes into the constituent plies of a component has, for the most part, been

[6] Huttrop, 1985.
[7] Vaccari, 1987, p. 88.
[8] Ibid., p. 89.
[9] Huttrop, 1985.
[10] Klein, 1987, pp. 94–95.
[11] Ibid., p. 97.

automated. In fact, every airframe manufacturer's plant we visited had one or more Gerber cutters.

This operation lends itself more readily to automation because every cut can be performed on a flat surface. Therefore, no consideration has to be made for curvatures and contours even though the plies may ultimately be laid up on very complex tools. The advantages are further enhanced if the individual plies tend to be large in area since manual operations can sometimes be limited by the reach, strength, and accuracy of human hands, depending upon ply design and fabric/tape width. Also if the plies are large, the automated machines spend less time stopping, starting, cutting, and changing directions. Ultimately, such two-dimensional operations as ply cutting are often considerably easier to automate than such three-dimensional operations as layup.

Ply management systems "instruct" the cutters on the marking and nesting of the material and is another area where efforts to automate are making progress. Once cut, plies are marked with information such as the material type, fiber orientation, sequence in layup, and out time. The plies are then collated and grouped into "trays" ready for layup onto the tool. Automated ply management systems are currently being implemented on the factory floor.

Other Areas. Net part operations, trimming, drilling, etc., are also making progress. Fully automated ultrasonic inspection procedures already exist. X-ray, acoustic emission, and nuclear magnetic resonance imaging are also being examined with respect to their potential for automation. Development of such postcure operations must consider the inherent damage tolerance of cured composite parts. As mentioned earlier, composites tend to be brittle and have low interlaminar properties. Therefore, they cannot tolerate mishandling as well as metallic parts during trim and drill operations.

Potential Savings. Meade, 1982, illustrates the potential savings from automation, particularly in layup, and the associated improved material properties. Estimates of fabrication time and cost were made for the composite stabilizer skins of a fighter aircraft using two procedures. The baseline case assumes predominantly manual operations, and the second case includes the effects of automation and advances in the raw material characteristics. Specifically, the second fabrication procedure included a robotic material transfer and orientation (layup) system, reusable vacuum bags, a low resin content prepreg material, and an automated tape dispensing system.

Comparisons were made between the largely manual procedures of the baseline case and the automated techniques used in the second case. The modifications to the operations are listed in Table 9 and the resultant cost reductions are illustrated in Fig. 14. The total fabrication hours required per shipset decreased from 35 to 10 hours. The largest single savings in this particular case was an 80 percent reduction (time) in the material transfer and orientation (layup) process. Savings were also achieved in debulking, pattern cutting, and bagging. The analysis also showed a 45 percent decrease in part cost (assuming 300 units) from the baseline case to the second case. And, although labor accounted for 45 percent of the cost in the baseline case, it accounted for only 17 percent in the second case. As the fabrication processes become more automated, the raw materials become the main cost driver.

Tool and material preparation times as well as curing and net part operations times were not affected in this case. However, these parameters would also have been affected if a thermoplastic material, instead of an improved (low resin content) thermosetting material, had been used in

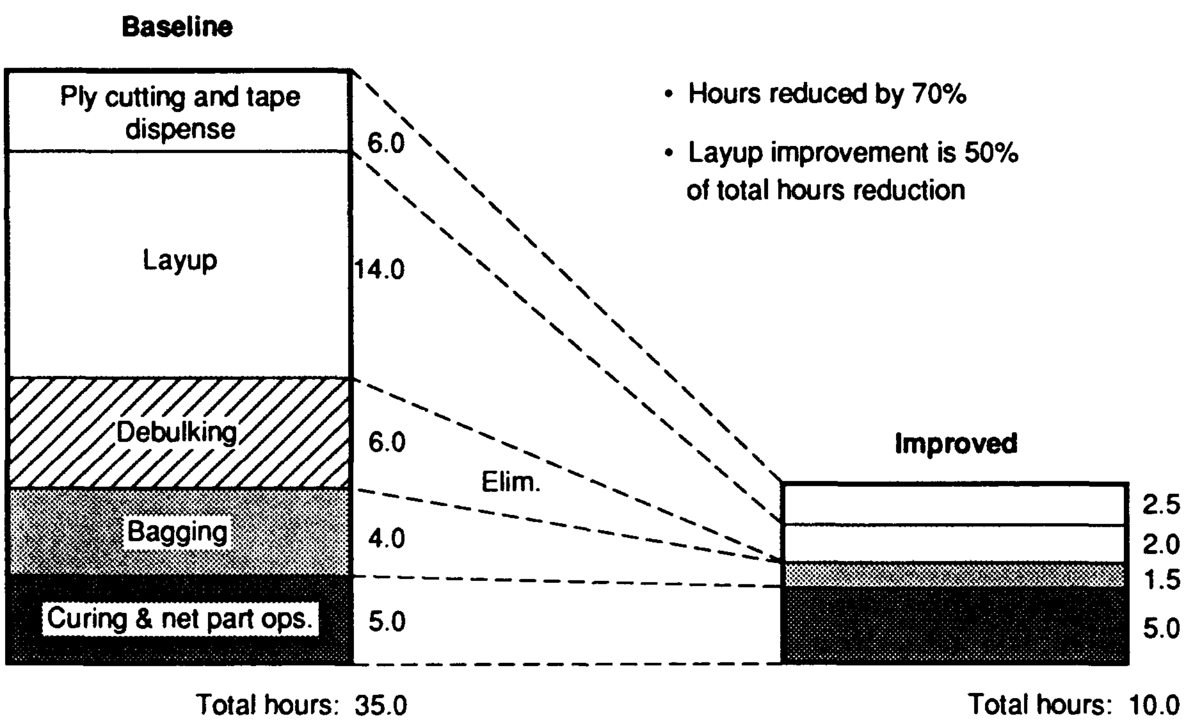
Table 9

MANUFACTURING COMPOSITE STABILIZER SKINS:
BASELINE AND IMPROVED OPERATIONS

Operation	Baseline	Improved
Material	60" wide	60" wide
Tape dispense	Manual	Automated
Ply cutting	Gerber knife	Gerber knife
Layup	Manual	Robot
Number of debulks	13	None
Bleeder plies	With	Without
Bag material	Disposable	Reusable
Postcure	With	Without[a]

SOURCE: Meade, 1982.

[a]Relaxation of production requirement.



SOURCE: Meade, 1982.

Fig. 14—Effect of improved operations on manufacturing hours of composite stabilizer skins

the second case. For example, freezers would not have been required to store the uncured prepreg material. The curing time would have dropped from hours to minutes; however, the autoclave requirements for temperature and pressure would have been more severe. A postcure would not have been required. Tool degradation would also have been aggravated by the shorter but hotter cure cycle.

A projection between manual and automated fabrication procedures was found for a fuselage section. This structure required the layup of inner skins, outer skins, longerons, and stiffeners. Total fabrication hours per shipset were projected to decrease from 630 to 230 hours. Total layup time decreased from 360 to 110 hours. Although total fabrication hours and total layup time both decreased by about 60 percent, layup costs still remain about 50 percent of the total using either method.[12]

Automation Conclusions. Introduction of automation and mechanization on the factory floor will be more evolutionary than revolutionary. As disussed above, certain aspects of manufacturing, such as ply cutting and management, are generally more amenable to automation than others, such as layup.

With respect to the cost of automation, we can safely say that any investment in capital equipment will require quantities sufficient to support it. Many companies' projections indicate future quantities will not warrant substantial levels of investment in automation (although moderate levels are planned). This is typified in LTV's decision to postpone construction on their Integrated Composites Center. Some respondents have indicated that automation in metal manufacturing will outpace automation of composites manufacturing, at least by 1995.

While companies have made great strides to automate composites manufacturing, automation is certainly not a given. Even though processes have been proven in development, the transition to production is not necessarily straightforward.[13] To enter a process into production there must be sufficient data and confidence and acceptable yields must be attained.

In summary, the extent of automation on the factory floor will be defined by at least some of the following constraints:

- Part sizes, contours, and complexities.
- Material selection.
- Quantities (low in this industry).
- Funds available for capital investment.

NEW METAL PROCESSES

Superplastic Forming/Diffusion Bonding

Superplastic forming and diffusion bonding (SPF and DB) are metal forming techniques. Both processes take advantage of the ability of some metals to endure elongations of several hundred percent under certain conditions (microstructure and process temperature) in order to both form (SPF) and bond together (DB) complex metallic shapes. The potential payoffs are reduced part and fastener count (unitized designs), excellent bond strength, and ultimately lower procurement and life cycle costs.

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[12]Marx, 1986, p. 401.

[13]The Beech Starship fuselage (500 lb) was originally conceived of as a single, filament-wound structure. The tooling for this approach cost about $6 million. Beech stated that they had difficulties achieving the desired levels of mechanical properties in the filament-wound composites and that the manufacturing cost was very high. Ultimately, they decided to hand-layup the fuselage in two pieces that are attached primarily by bonded surfaces.

The generic process of superplastic forming consists of placing flat sheet stock (titanium, aluminum) over a die of the desired part shape. The stock is heated to the processing temperature (exceeding 1,500°F) and a burst of inert gas forces the flowing material into the die. The metal stock assumes the required part shape and is held under temperature and pressure for a short time before cooling.

Diffusion bonding requires that the mating surfaces be pressed together with pressures that locally exceed the yield stress of the material. The bonds result from the diffusion of atoms across the mating surfaces at elevated temperatures. The strength of these bonds approaches the strength of the parent metal. Not all metals may be diffusion bonded (aluminum cannot). Examples of diffusion bonded titanium parts include the wing carry through of the B-1B bomber and the space shuttle main engine mountings.

SPF and DB processes require very similar processing environments and can be performed sequentially (with the same equipment); two or three parts may be first superplastically formed and then bonded together. Since the processes are so compatible, considerable reductions in manufacturing time and costs are possible.

SPF/DB are primarily used to manufacture titanium parts. Titanium is nearly indispensable in airframe applications for several reasons: high temperature capability (1,000°F), low density, high strength and stiffness, high damage tolerance, and corrosion resistance. Titanium is difficult to machine, but its microstructure is adaptable to SPF/DB processes, thus avoiding much expensive machining processes. Research is also being performed on SPF aluminum and aluminum-lithium.

SPF/DB titanium parts have demonstrated 10–50 percent weight savings, but they cannot compete with the cost of conventional aluminum parts. Raw material costs alone exceed aluminum by a factor of 8. Therefore, for the foreseeable future, titanium SPF/DB parts will be confined to those areas requiring exceptional performance.

There are many examples of SPF/DB demonstration components (usually related to engines), but only one production part was found: a nacelle beam frame for the B-1B. With conventional techniques, this frame has eight detail parts and 96 fasteners. With the SPF/DB processes, these numbers were reduced to one detail part and zero fasteners. Structural weight was reduced by 30 percent. SPF/DB is readily adaptable for producing reinforced sheet structures, internally stiffened structures (two sheets), and sandwich structures (three sheets).

Powder Metallurgy

Powder metallurgy procedures use, as the name implies, very rapid solidification rates of powder forms. Hot isostatic pressing (HIP) is the consolidation technique of primary interest with respect to airframe parts. The P/M process for such structures can be broken down, very roughly, into three operations:[14]

- Powder production.
- Containerization.
- Hot isostatic pressing.

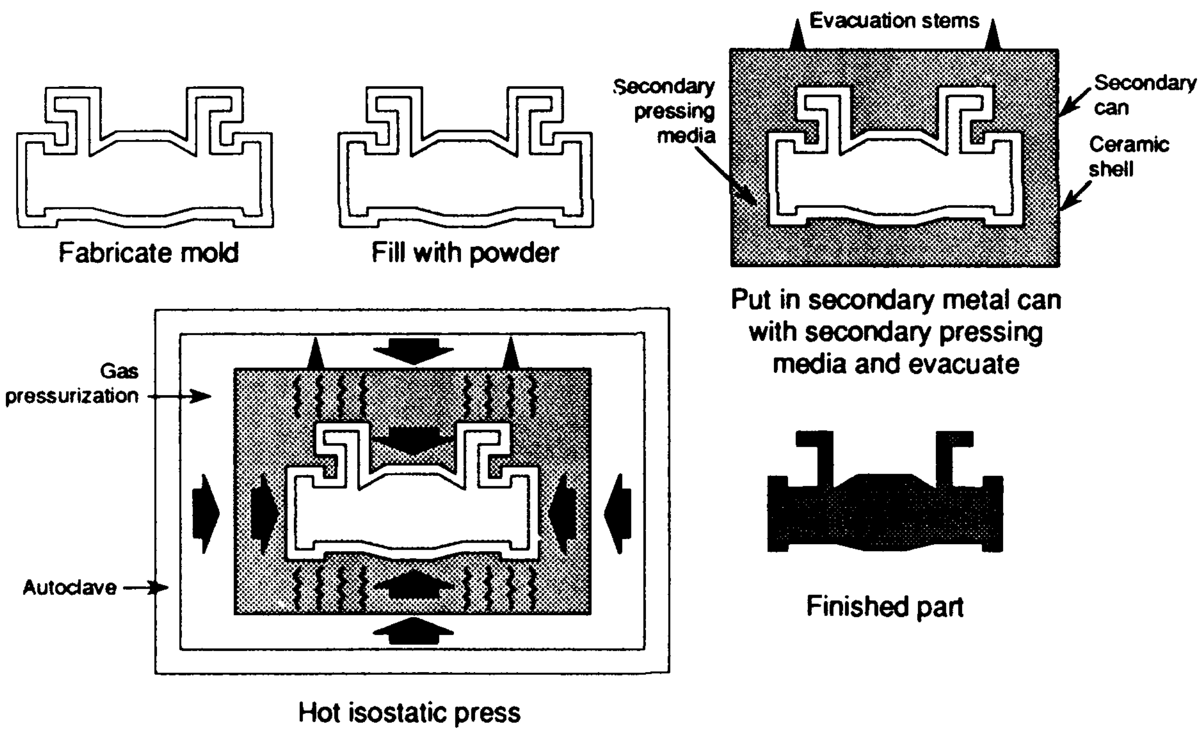
The first step is to produce "clean spherical powder free of contaminants." Vacuum induction melting followed by inert gas atomization is a common process. The second step consists of loading the powder into metallic or ceramic molds that define the shape of the part (including any desired tolerances). The smaller the tolerance envelope is, the more efficient, or

[14]Dulis, 1986.

near net shape, the operation becomes. The production of these molds can be a complex matter and there is the possibility of introducing contaminants from the mold into the part. The mold is then placed inside a steel container; any remaining volume in the container is filled with a granular ceramic medium that transfers external pressure to the mold and part. The steel container is then welded, outgassed, and sealed (see Fig. 15). The third step, which is the HIP phase, consists of the application of a specified cycle of heat and pressure. Cycle times can surpass 8 hours, maximum temperatures and pressures can exceed 2,000°F and 25,000 psi (see Fig. 16).

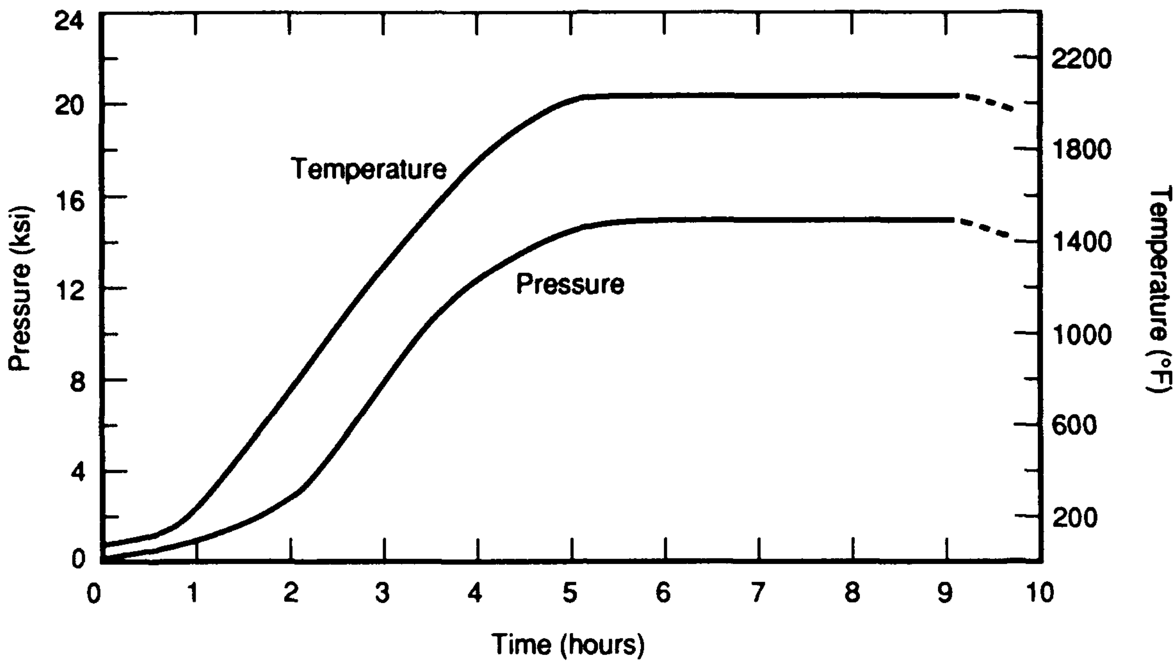
Further development of P/M processes must occur in several areas:

- Better analytical modeling and quantitative descriptions of the processes are required.
- Sensors that provide feedback on key microstructural parameters of the part need to be developed and incorporated into manufacturing procedures.
- Nonmetallic inclusions that are introduced in the manufacturing procedures can substantially reduce the part's low-cycle fatigue (LCF) properties. There has been considerable concern about the reliability of LCF properties in P/M parts. Some speculation about the cause of an F-18 crash in 1980 centers around the fatigue life of some of the P/M engine components. The true cause remains unknown. LCF properties in P/M parts can be very good when adequate attention is paid to the manufacturing



SOURCE: E.J. Dulis, "Near Net Shape Process Using HIP of Alloy Powder Particles," Net Shape Technology in Aeronautical Structures, Vol. III, National Academy Press, 1986.

Fig. 15—The ceramic mold process for producing complex shapes from powder



SOURCE: E.J. Dulis, "Near Net Shape Process Using HIP of Alloy Powder Particles," Net Shape Technology in Aeronautical Structures, Vol. III, National Academy Press, 1986.

Fig. 16—A typical superalloy hot isostatic process cycle

procedures, but verification tests should be performed. This problem may diminish as nonceramic processes evolve.

- According to one paper, considerable emphasis has been placed on developing the strength and environmental resistance of the alloys in P/M parts while the metalworking needs of the fabricator have been ignored.[15] The restrictions on metalworking temperatures may inhibit the ability of the alloys to undergo superplastic deformations and thereby negate the possibilities of many efficient net shape procedures.

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[15]Dulis, 1986.

IV. COST INFORMATION

Costs by material type are presented and discussed in this section. These results were derived from worksheets distributed to the airframe industry. After the initial reporting phase of the project, participants were asked to verify that their response was consistent with our format and to respond to a series of questions clarifying the results. This additional information was incorporated into the report.

INDUSTRY WORKSHEETS

We considered an industry survey to be the best approach to collect cost information because (1) the number of historical data points is limited, (2) we were interested in materials not yet incorporated into production aircraft, and (3) we wished to capture the increased level of usage in aircraft. Worksheets were designed to provide information that was material sensitive and could be used to build up to an estimate of total airframe structure. A list of the participating companies is provided in Table 10.

Information was requested for the materials listed in Table 11 for late-1980 and mid-1990 time periods. In general, we found that companies were willing to provide cost information for materials with which they had actual production experience. They were reluctant, however, to provide projections for materials still in the developmental stage because of the sensitivity and uncertainty of the data. Consequently, the list of materials for which we actually obtained sufficient data is considerably smaller:

Conventional aluminum
Aluminum-lithium
Conventional titanium
Conventional steel
Graphite/epoxy
Graphite/bismaleimide
Graphite/thermoplastic

Nevertheless, despite its reduced scope, the list still encompasses the composite materials currently considered most important to the next generation of aircraft: graphite/epoxy, graphite/ bismaleimide, and graphite/thermoplastic.

Table 10

STUDY PARTICIPANTS

Boeing Military Airplane Company	LTV Aerospace and Defense Aircraft Group
General Dynamics Corporation	McDonnell Douglas Corporation
Grumman Aerospace Corporation	Northrop Aircraft Division
Lockheed Aerospace Systems Corporation—California Division	Rockwell International Group
Lockheed Aerospace Systems Corporation—Georgia Division	

Table 11

STRUCTURAL MATERIALS LIST

Composite Materials
Polymer matrix
epoxy
graphite/epoxy
Kevlar/epoxy
boron/epoxy
polyimide
graphite/bismaleimide
glass/bismaleimide
thermoplastic
graphite/thermoplastic
Metal matrix
graphite/aluminum
boron/aluminum
Ceramic matrix
Carbon/carbon
Whisker filled composites
aluminum with silicon carbide whiskers
Glass fiber-reinforced plastic
Advanced Metals and Alloys
Aluminum
conventional aluminum
aluminum-lithium alloys
powder metallurgy aluminum
Titanium
conventional titanium
powder metallurgy titanium
TiAl and Ti3Al alloys
Steel
conventional steel

The worksheet[1] consisted of five sections categorized as follows:

I. Corporate material experience
II. Material usage
III. Technical data
IV. Cost data
V. General questions

The corporate experience section requested previous and planned experience with advanced materials in major subassemblies. Section II requested information on where a material might be incorporated into an airframe (wing, fuselage, or empennage). The third section requested background information on material characteristics and the aircraft application. For example, the work sheet included questions on material strength and stiffness, material form (tape or broadgood), type of aircraft and its performance (subsonic or supersonic), the design concept,

[1]A sample worksheet is provided in App. A.

and level of automation in manufacturing. The purpose of this section was to set the stage for subsequent cost-related questions.

Section IV requested cost data for each of the functional elements listed in Table 1[2] as well as information concerning cost improvement curves, weight sizing factors, and material buy-to-fly ratios. The last section of the worksheet contained questions on funding profiles, tooling concepts, etc.

Generally speaking, responses for the late-1980s time period reflect a company's cumulative production experience with a given material type through that time. In contrast, responses for the mid-1990s reflect a company's projection of what costs for these materials will be based on: ongoing development work and an assumed level of capital investment. Unfortunately, when the worksheets were distributed in 1987, the outlook for the defense industry was considerably brighter than is the current case. Given current production stretchouts and cancellations, it is not clear that companies will be willing (or able) to make the capital investments necessary to achieve the projections. Consequently, the mid-1990 values are undoubtedly on the optimistic side.

UNDERLYING ASSUMPTIONS

1. All costs are in FY90 dollars.
2. The labor hours and material cost factors include all structural fabrication and all assembly up through the airframe group level (wing, fuselage, and empennage). The values do not include final assembly and checkout nor any subsystem installation done before final assembly and checkout.
3. All labor hour and material cost factors represent cumulative average values for a quantity of 100 aircraft and a finished material weight of 1000 lb.[3] We used the log-linear cumulative average form because of its (1) simplicity, particularly in a program's planning stages or when one is dealing with large numbers of aircraft, and (2) compatibility with a previous RAND methodology.[4]
4. All responses assumed some mix of material form. The form used will vary from part to part and will depend on a specific part's contour or strength and stiffness requirements. In an aircraft many parts will require broadgoods, tape, or both.
5. Most responses assumed cocuring of smaller parts and mechanical fastening of large substructures.
6. All responses assumed some mix between manual and semi-automated production facilities (for composites this is characterized somewhat in Sec. II). No one assumed a completely automated system either in the late 1980s or the mid-1990s.

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[2]Nonrecurring engineering, nonrecurring tooling, recurring engineering, recurring tooling, recurring manufacturing labor, recurring manufacturing material, and recurring quality assurance. Definitions of these cost elements appear in App. C.

[3]All recurring cost values obtained as part of the industry survey represented *cumulative average* hours (or dollars) per pound for 100 units. Not all the values reflected the same finished material weight. Consequently, we adjusted each contractor's values to a finished material weight of 1000 lb using the weight-sizing factors suggested by that contractor.

[4]Hess and Romanoff, 1987. The log-linear unit curve formulation may be more appropriate when one is dealing with small quantities or segmented curves, or when actuals on earlier lots are available.

SUMMARY OF WORKSHEET RESPONSES

This section presents the results of our industry survey beginning with the nonrecurring elements and followed by the recurring cost elements. For each material type and time period, both the average and range of company responses are reported. We determined the average values for each material type by averaging each firm's minimum and maximum values and then averaging across firms. The range of values presented for a given material type represents the minimum and maximum values reported regardless of firm. Ideally, the range of intercompany responses would be fairly narrow. Unfortunately, this is frequently not the case. However, plausible explanations for such intercompany differences do exist:

- *Type of aircraft produced by company.* Generally speaking, combat aircraft involve more complex parts and utilize higher-temperature materials than noncombat aircraft. Companies producing combat aircraft will tend to have higher per-pound costs than producers of noncombat aircraft.
- *Corporate experience with specific material types.* It is not unreasonable to expect that as a firm gains production experience with a given material type, its costs for that material will fall. Firms with limited new material production experience are likely to have higher than average costs.
- *Level of factory automation.* Everything else being equal, firms with higher levels of automation are likely to have lower per-unit labor charges (but higher per-unit overhead charges).

Nonrecurring Cost Elements

Since nonrecurring costs are often a function of specific program characteristics, simple ratios for an all-aluminum aircraft were requested on the worksheets distributed to industry, particularly for nonrecurring engineering and nonrecurring tooling. The ratios reflect the nonrecurring effort required for a component made of a particular material relative to the effort that would be expended for that same component to be made of late-1980s aluminum baseline. These factors apply to airframe structure design and exclude subsystem development.

Nonrecurring Engineering. Nonrecurring engineering includes the engineering hours spent designing the airframe. Specifically it includes hours expended for (1) design, consisting of studies, stress analysis, aerodynamics, weight and balance analyses, and integration; (2) wind-tunnel models and mockups; (3) laboratory testing of components, subsystems, and static and fatigue articles; (4) preparation and release of drawings; and (5) process and materials specification. Excluded are engineering hours not directly attributable to the airframe: flight testing, ground handling equipment, spares, and training equipment. The industry average, minimum, and maximum ratios for nonrecurring engineering hours per pound are presented in Table 12.

On average, nonrecurring engineering hours per pound in the late 1980s are estimated to be 40 to 70 percent higher for composites than for metals. Only one response suggested the effort required would be less. The specific reasons given for additional nonrecurring engineering hours for composite materials are:

- Additional effort is required to manage the individual composite plies. Composite designers have to specify the number and orientation of the plies for each part to achieve the desired strength and stiffness characteristics.

Table 12

NONRECURRING ENGINEERING HOURS PER POUND RATIOS
(Late 1980s, aluminum = 1.00)

Material Type	Late 1980s Average	Late 1980s Min/Max	Mid-1990s Average	Mid-1990s Min/Max
Aluminum	1.0	1.0/1.0	1.0	0.8/1.0
Al-lithium	1.1	1.0/1.3	1.0	0.9/1.3
Titanium	1.1	1.0/1.3	1.0	0.9/1.3
Steel	1.1	0.9/1.3	1.1	0.9/1.3
Graphite/epoxy	1.4	0.9/2.5	1.2	0.7/2.0
Graphite/bismaleimide	1.5	0.9/2.5	1.3	0.7/2.0
Graphite/thermoplastic	1.7	0.9/3.0	1.4	0.7/2.5

- Engineers are not familiar with the materials and have little experience designing with them. Additional effort is required to develop the analytic tools necessary to model and test them.
- There is a lack of material standards and universally accepted safety margins since many properties of these materials are unknown or unproven. New material properties have much greater variability than metals. Therefore, additional modeling of material properties and behavior under load conditions is required.
- Achieving high tolerances when required by performance considerations is difficult.

The main argument given for fewer engineering hours is:

- Design unitization will reduce the part count and simplify the overall design process.

One company suggested that in total, engineering hours are insensitive to material type. That is, a specific part or subassembly will take X amount of engineering hours regardless of the material composition. However, because composites weigh less, a factor must be applied to the hours per pound to compensate. This is in fact the approach NAVAIR took in their study on composites, which relates initial engineering hours/pound to weight savings.[5] Presumably greater levels of weight savings will be achieved either by strict design controls and novel techniques or increased use of the new materials. The NAVAIR factors for nonrecurring engineering are a function of estimated weight savings. For a weight savings of 10 percent, engineering hours/pound would increase 11 percent; for a 20 percent savings, hours would increase by 25 percent; and for a 30 percent savings in weight, hours would increase by 43 percent (or 1/(1-.30)).

Overall, engineering hours are expected to decrease between the late 1980s and the mid-1990s. For the metals, the decrease was on the order of 10 percent and attributable primarily to the greater use of computer aided design. For the composites, decreases were estimated to be slightly greater—between 10 percent and 25 percent—because of the combined effect of increased computer aided design and greater familiarity with the materials.

Nonrecurring Tooling. Tooling refers to the tools designed solely for use on a particular airframe program, and includes layup tools, autoclave tools, assembly tools, dies, jigs,

[5]Weathersbee, 1983.

fixtures, work platforms, and test and checkout equipment. General purpose tools or machinery—such as Gerber cutters, milling machines, presses, routers, lathes, X-ray and ultrasonic equipment, and autoclaves—are considered capital equipment. Tooling hours include all effort expended in tool and production planning, design, fabrication, assembly, installation, modification, maintenance, rework, and programming and preparation of tapes for numerically controlled machines. Nonrecurring tooling refers to the costs of the initial set of tools and all duplicate tools produced to attain a specified production rate.

Industry ratios for nonrecurring tooling hours per pound are presented in Table 13. Nonrecurring tooling for composite products is, and will be, substantially higher than nonrecurring tooling for aluminum products. The reasons given to support this conclusion are:

- Exposure to high temperatures and pressures in the autoclave.
  - Increased tool design effort. For example, in the autoclave the tools will expand because of exposure to heat. Tool designers must consider the relationship of the coefficients of thermal expansion between the tool and the material being processed to ensure the desired final part shape is attained.
  - Higher cost tools. The tools will have to be able to withstand the high temperatures and pressures of the autoclave, thereby limiting the use of aluminum tools and requiring the use of steel, graphite, and electroplated nickel tools. In general, finished tools made of these materials are more expensive than those made of aluminum.
- The need for highly accurate tools since finished parts can not be reworked once autoclaved, as is possible with metals.
- The need for more tools because of the long processing times in the autoclave. Some have said that tools will now be required to make the production tools (if the tools themselves are made of composites).

One respondent believes tooling costs may be decreased if a sufficiently unitized design can reduce the overall quantity of tools required. Although individual tools may be very expensive, a unitized design will mean fewer parts and smaller tool quantities.

The differences in nonrecurring tooling costs among the three composites are largely attributable to differences in processing temperature. Currently, processing temperatures above 625° to 650°F are difficult to manage. It was estimated that processing temperatures

Table 13

NONRECURRING TOOLING HOURS PER POUND RATIOS
(Late 1980s, aluminum = 1.00)

	Late 1980s		Mid-1990s	
Material Type	Average	Min/Max	Average	Min/Max
Aluminum	1.0	0.9/1.0	1.0	0.9/1.0
Al-lithium	1.2	1.0/1.7	1.1	0.9/1.7
Titanium	1.4	0.9/3.7	1.4	0.9/3.4
Steel	1.1	1.0/1.4	1.1	1.0/1.4
Graphite/epoxy	1.6	0.7/2.5	1.4	0.5/2.0
Graphite/bismaleimide	1.7	0.7/2.5	1.5	0.5/2.3
Graphite/thermoplastic	2.0	0.7/3.0	1.6	0.5/2.5

above this threshold can add 50 percent to the cost of an individual tool. Newer composites such as polyimides and thermoplastics require processing well above this mark.

Tooling costs are expected to decrease by the mid-1990s. One reason given was that computer aided design will help in the planning of the tools. Better material standards, familiarity with the manufacturing processes, and improvements therein are other likely reasons for the decrease. In the case of titanium, one company believed tooling costs would rise in the 1990s if SPF/DB processes are used.

The previous discussion and the cost information presented centered on tooling costs per pound for the airframe. From part to part, tooling costs can vary considerably based on contour complexity, number of attachment points, tolerance requirements, autoclave processing temperature and pressure requirements, and size.

Nonrecurring Cost Elements Summary. Indications are that nonrecurring costs for airframe structure incorporating new materials will be higher than those for a conventional airframe, in some cases much higher. As expertise is gained and historical data on new material properties increase, these values will decrease but probably not to the levels of conventional materials.

Recurring Cost Elements

This section provides cost-per-pound factors for each material type for each of the five recurring cost categories: engineering, tooling, manufacturing labor, manufacturing materials, and quality assurance. Once again, both the average and range of company responses are presented. Unfortunately, the range of responses for the recurring elements is frequently very large. However, there was a tendency for the majority of responses to cluster in narrow ranges. Consequently, we also document the cluster range and recommend its use for general sensitivity analyses rather than the larger, unrestricted ranges.

Recurring Engineering. Industry responses for recurring (or sustaining) engineering are summarized in Table 14. Recurring engineering clearly varies by material type. Unfortunately, we have not yet been able to establish a logical relationship between material type and sustaining engineering. In fact, two companies indicated that new materials will not affect sustaining engineering. Moreover, a third company declined to provide any response for this cost category, stating that sustaining engineering hours are a function of many variables (government mandates, funding levels, program phase, and aircraft mission), none of which are related to material type.

In any case, for the firms that did respond, sustaining engineering hours per pound for graphite/epoxy are expected to be twice the hours of aluminum in both the late 1980s and mid-1990s (with graphite/bismaleimide and graphite/thermoplastic somewhat higher). By the mid-1990s, graphite/epoxy is projected to be roughly twice as expensive per pound as aluminum; however, the bismaleimides' and thermoplastics' sustaining engineering hours per pound will be roughly equivalent to those of the epoxies.

Recurring Tooling. Recurring (or sustaining) tooling refers to the effort required to maintain and repair production tools. These hours are reported to be greater for structure fabricated with new materials than with aluminum. The values shown in Table 15 are over twice those of aluminum.[6] Some of the reasons for higher recurring tooling costs for composite materials are:

[6]A NAVAIR study (Weathersbee, 1983) reported approximately a 30 percent increase in sustaining tooling for graphite/epoxy, less than the values shown here.

Table 14

RECURRING ENGINEERING HOURS/POUND
(Cumulative average hours for 100 units for 1000 lb of structure)

Time Period/ Material Type	Average Value	Min/Max Value in Cluster	Min/Max Value in Sample
Late 1980s			
Aluminum	1.0	0.4/1.0	0.4/2.3
Al-lithium	1.1	0.5/1.1	0.4/2.5
Titanium	1.4	0.7/1.5	0.6/2.3
Steel	1.1	0.6/1.0	0.5/2.3
Graphite/epoxy	1.9	0.8/2.5	0.4/4.2
Graphite/bismaleimide	2.1	1.2/2.8	0.4/4.5
Graphite/thermoplastic	2.9	0.9/3.2	0.6/7.5
Mid-1990s			
Aluminum	0.9	0.3/1.0	0.3/2.1
Al-lithium	1.0	0.4/1.0	0.4/2.2
Titanium	1.2	0.7/1.5	0.6/2.1
Steel	1.1	0.5/1.0	0.5/2.3
Graphite/epoxy	1.5	0.6/1.9	0.3/3.6
Graphite/bismaleimide	1.6	1.0/2.3	0.3/3.6
Graphite/thermoplastic	1.4	0.6/2.2	0.3/3.6

Table 15

RECURRING TOOLING HOURS/POUND
(Cumulative average hours for 100 units for 1000 lb of structure)

Time Period/ Material Type	Average Value	Min/Max Value in Cluster	Min/Max Value in Sample
Late 1980s			
Aluminum	1.6	0.6/1.7	0.3/ 5.2
Al-lithium	1.7	0.5/1.9	0.3/ 4.6
Titanium	3.0	0.5/2.9	0.5/ 9.7
Steel	2.3	0.6/2.5	0.4/ 7.3
Graphite/epoxy	3.6	0.6/6.7	0.6/ 9.3
Graphite/bismaleimide	3.7	0.6/6.8	0.6/ 9.3
Graphite/thermoplastic	3.9	0.7/7.1	0.7/10.5
Mid-1990s			
Aluminum	1.5	0.5/1.7	0.3/ 4.5
Al-lithium	1.7	0.5/1.9	0.3/ 4.6
Titanium	2.6	0.6/2.8	0.5/ 8.2
Steel	2.3	0.6/2.3	0.4/ 7.3
Graphite/epoxy	3.2	0.4/6.0	0.4/ 8.6
Graphite/bismaleimide	3.3	0.5/6.1	0.4/ 8.5
Graphite/thermoplastic	3.8	0.6/7.0	0.4/10.5

- Tools must be cleaned after going through the autoclave, a time consuming process.
- Because of thermal cycling in the autoclave, tools undergo more wear and need to be replaced more frequently.
- Composite tools are less durable than metal tools since nicks and scratches cannot necessarily be machined away.
- Storing composite tools is more demanding.

The graphite/bismaleimide and graphite/thermoplastic values in Table 15 reflect the slightly higher temperatures and longer cycles required for an autoclave cure over those required to process graphite/epoxy. If manufacturing processes that replace autoclave cure are developed for thermoplastics (those taking advantage of thermoplastics' reformability, such as the stamping out of parts), some respondents indicated that tooling hours may actually decrease below the levels reported for graphite/epoxy.

Recurring Manufacturing. Recurring manufacturing hours/pound are presented in Table 16. In general, the minimum values are based on larger, fairly simple subsonic aircraft, whereas the maximum values are based on aircraft utilizing more complex, contoured structures.

Manufacturing hours per pound clearly vary with material type. Aluminum-lithium and steel are only slightly more expensive on an hours per pound basis than aluminum, while titanium is noticeably more expensive, because titanium is particularly difficult to machine and is highly resistant to chemical milling. Finally, composites are much more expensive than aluminum on a dollar per pound basis. Graphite/epoxy and graphite/bismaleimide are approximately twice as expensive as aluminum in the late 1980s. Informed judgments voiced during our visits were that graphite/bismaleimide requires 10 percent to 20 percent more hours per pound than

Table 16

RECURRING MANUFACTURING HOURS/POUND
(Cumulative average hours for 100 units for 1000 lb of structure)

Time Period/ Material Type	Average Value	Min/Max Value in Cluster	Min/Max Value in Sample
Late 1980s			
Aluminum	9.1	7/12	3.5/12.6
Al-lithium	10.4	8/13	4.0/15.6
Titanium	14.8	8/17	6.7/30.8
Steel	10.9	8/17	1.5/22.6
Graphite/epoxy	16.2	10/20	2.8/40.8
Graphite/bismaleimide	19.1	9/24	5.0/46.9
Graphite/thermoplastic	16.8	10/25	5.0/28.6
Mid-1990s			
Aluminum	8.0	6/10	3.5/10.9
Al-lithium	9.4	7/12	3.7/13.6
Titanium	12.6	7/15	5.8/26.8
Steel	10.9	8/15	1.3/19.6
Graphite/epoxy	13.3	9/16	2.8/33.4
Graphite/bismaleimide	16.5	8/20	4.0/40.3
Graphite/thermoplastic	14.5	9/21	4.2/25.5

graphite/epoxy because (1) it has less tack and poorer drapability characteristics, making it more difficult to lay up; and (2) it has a slightly longer cure time.

Although many believe thermoplastics have the potential to be treated more like metals—stamped, pultruded, welded, and machined easier; reformed; processed quicker; etc.—few companies have experience working with them, and the processes for manufacturing them are still in development. Other studies have estimated thermoplastic materials hours per pound to be twice those of aluminum.[7] We have been told that if thermoplastics are processed as thermosets (laid up and autoclaved), then they will never be cost-effective. Laying up thermoplastics is more difficult because they have no tack and poor drapability characteristics at room temperature, necessitating external heat. If autoclaved, thermoplastics cure temperature requirements are at the limits of the existing capability in most manufacturing facilities.[8] Achieving the values reported here appear to be contingent on the successful development of processing technologies for thermoplastics and the willingness to invest in the capital equipment necessary to use them.

Every company that did projections assumed automation would reduce manufacturing hours per pound to some extent. Some believe that in mass production, with automation, graphite/bismaleimide will be indistinguishable from graphite/epoxy in terms of manufacturing hours per pound.

Examples taken from the *DOD/NASA Structural Composites Fabrication Guide* and reported in Sec. II outline possible improvements in the process that may greatly reduce manufacturing hours. The *Guide* states that manual layup accounts for 50 percent of a spar's labor cost and 90 percent of a cover's labor cost. Industry's ability to develop and incorporate automated techniques, with a special emphasis on potential payoffs to automating layup, will determine the amount of savings achievable. The *Guide* also states that as much as half of the total fabrication cost of structural composite torque boxes can be attributed to mechanical fastening operations.[9] It follows that another essential ingredient to reducing costs and weight is greater use of cocured assemblies.

It is difficult to separate the effects of low observability requirements from the material effects on hours per pound. Some companies believe that this requirement could add 15 percent to 30 percent to recurring manufacturing hours (some of the higher values in our range as shown in Table 16 may reflect this complexity).

Recurring Quality Assurance. Industry average, minimum, and maximum recurring quality assurance hours per pound are shown in Table 17. Some believe quality assurance hours across all material types will be higher in general because of the customer requirement for an improved product. A NAVAIR study did not quantify but stated that quality assurance hours for composites will be much higher than those for aluminum. Because the materials are new and unproven, more testing is required. Also, composite materials quality assurance hours may be higher because of added requirements driven by the materials themselves and their associated processes. For example, these materials must be inspected for internal delaminations and bond integrity, both of which are not possible to do visually and neither of which is a concern in metal fabrication. In addition, test results must be well characterized so that engineering decisions with respect to defect severity and repair (or scrap) procedures can be made. To compound these problems, testing procedures and guidelines are still evolving. Finally, the type and extent of the quality assurance procedure used will depend on the

[7]Pingel, 1987.

[8]Tooling would also be affected.

[9]Meade, 1982.

Table 17

RECURRING QUALITY ASSURANCE HOURS/POUND
(Cumulative average hours for 100 units for 1000 lb of structure)

Time Period/ Material Type	Average Value	Min/Max Value in Cluster	Min/Max Value in Sample
Late 1980s			
Aluminum	1.7	0.7/ 2.7	0.3/ 3.8
Al-lithium	1.8	0.8/ 2.8	0.4/ 3.5
Titanium	2.7	1.0/ 4.4	0.5/ 6.0
Steel	2.4	0.9/ 3.9	0.5/ 5.3
Graphite/epoxy	4.1	0.8/ 7.4	0.7/10.9
Graphite/bismaleimide	4.3	0.8/ 7.8	0.8/11.8
Graphite/thermoplastic	4.4	1.0/ 7.8	0.8/10.6
Mid-1990s			
Aluminum	1.5	0.6/ 2.4	0.3/ 3.3
Al-lithium	1.7	0.8/ 2.6	0.4/ 3.3
Titanium	2.4	0.9/ 3.9	0.5/ 5.2
Steel	2.4	0.9/ 3.9	0.5/ 5.3
Graphite/epoxy	3.1	0.5/ 5.8	0.5/ 9.2
Graphite/bismaleimide	3.6	0.6/ 6.6	0.6/10.4
Graphite/thermoplastic	3.4	0.7/ 6.1	0.6/ 9.1

criticality of the component to the overall system. Some components may require several different tests. The values reported here reflect composite usage between 30 and 50 percent of structural weight, some of which will occur in structural and load-bearing components.

Material Costs. Three elements determine total material cost: raw material cost, buy-to-fly ratio, and material burden rate. The buy-to-fly ratio is the amount of material purchased to complete a pound of finished part that "flies" away. Since in our context buy-to-fly ratios are used to calculate the total material bill, they should include material lost in poor material handling processes, the machining or cutting processes, the fabrication of parts that are eventually *scrapped* because of flaws or deficiencies, and other steps in the manufacturing process that cause material to be lost.[10]

The costs reported here represent a particular material's average dollars per pound for the entire airframe structure, implicitly incorporating a typical mix of material forms. A material's specific costs and buy-to-fly ratio may vary widely depending on the form (sheet, plate, billet, tape, broadgood) or precise grade of material. For part level analysis, other detailed information regarding the specific material form would be more appropriate.[11]

The material burden rate captures the costs of purchasing, handling, and inspection. The rates reported to us varied from company to company, reflecting differing accounting

[10]The manufacturing labor improvement curve slope may also be affected by the scrap rate. We have been told that during the very early stages of production, scrap rates as high as 200 percent are not uncommon. Once production is entered, however, these problems will have been resolved and parts will only rarely be discarded. Therefore, early manufacturing labor improvement curve slopes may be steeper than those experienced later in production.

[11]For example, limited information suggests that buy-to-fly ratios for composite broadgoods are 25 to 75 percent higher than those for composite tape. Based on this limited information, the values in Table 18 more closely resemble composite tape ratios. Buy-to-fly ratios can also be a function of how close the raw material is to the final form. A lower buy-to-fly ratio should correspond to higher material costs since the material would have required additional processing (to get it close to final form).

procedures. We have been informed these procedures are being revised to ensure burden rates uniformly represent value added. The industry average of 15 percent was used in subsequent calculations (the range was 0 to 33 percent).

Table 18 summarizes the information we have received on raw material costs and buy-to-fly ratios. Composite raw material costs are expected to decrease in the mid-1990s approximately 20 percent because of greater supplier economies of scale. Buy-to-fly ratios are also estimated to decrease. At the time these projections were made, the mid- to late 1980s, industry expected that the demand for composite materials was increasing and therefore economy-of-scale benefits would be realized. However, the current situation is less certain with the potential cancellation or stretchout of major aircraft programs likely to reduce projected benefits. Additionally, increased energy prices may adversely affect petroleum based materials' costs.

Advanced material costs have been of great concern to manufacturers because they are so much more expensive than traditional materials. Some of these additional costs have been offset by lower buy-to-fly ratios. Moreover, material costs constitute a small proportion of overall recurring dollars (in the range of 2 to 9 percent as shown in Tables 21 and 22). Therefore, even though composite materials can cost an order of magnitude more than stock metal material, the effect on total cost is small.

Cost-Improvement Curve Slopes and Weight-Sizing Factors. Suggested cost-improvement curve slopes for adjusting the recurring cost values to quantities other than 100

Table 18

MATERIAL COST FACTORS

Time Period/ Material Type	Buy-to-Fly Ratio	Raw Material $/lb (FY90$)	Material[a] $/lb (FY90$)
Late 1980s			
Aluminum	2.5	11	27
Al-lithium	4.2	17	72
Titanium	3.0	26	76
Steel	2.1	8	18
Graphite/epoxy	1.9	69	130
Graphite/bismaleimide	1.9	78	146
Graphite/thermoplastic	1.9	91	173
Mid-1990s			
Aluminum	2.2	10	22
Al-lithium	2.7	9	25
Titanium	3.0	24	72
Steel	2.1	8	18
Graphite/epoxy	1.8	57	102
Graphite/bismaleimide	1.8	61	111
Graphite/thermoplastic	1.8	66	119

[a]No burden added. The average industry burden is 15 percent.

are provided in Table 19 along with weight-sizing factors for adjusting finished-material weights to values other than 1000 lb.[12]

Recurring Cost Element Summary. As the data in Tables 14 through 18 show, all recurring cost elements are expected to be greater for composite materials than for metals. The support categories are higher for composites than for metals and the manufacturing hours are greater as well. Some reduction of all values in the 1990s is expected. Automation, general improvement of the processes, and greater material supply are expected to be the important factors in the reduction of advanced materials' total recurring costs per pound. Weight savings achieved through the use of these new materials will help mitigate the additional costs.

APPLYING THE DATA

This section develops estimates of total recurring dollars per pound by material type and time period. Industry average labor hours are converted to dollars by means of the fully burdened labor rates shown in Table 20.[13] Material costs are then burdened and added to labor costs to get an overall recurring dollars per pound. The results are shown for the late 1980s and mid-1990s in Tables 21 and 22, respectively. The following conclusions can be drawn from these two tables:

Table 19

COST-IMPROVEMENT CURVE SLOPES AND WEIGHT SIZING FACTORS
(Percent)

	Average	Minimum	Maximum
Improvement curve slopes[a]			
Recurring engineering	70	58	85
Recurring tooling	73	65	91
Manufacturing labor	78	71	91
Manufacturing material	87	70	97
Quality assurance	78	63	100
Weight sizing factors[b]			
Labor	85	80	90
Material	96	85	100

[a]RAND Aircraft Airframe Database (proprietary database consisting of cost data obtained from major aircraft firms, either directly from their records or indirectly through standard Department of Defense reports such as the Contractor Cost Data Reporting (CCDR) system).

[b]Composites industry survey (see App. A, Worksheet IV).

[12]As part of the industry survey, we requested improvement curve slopes by material type and time-frame for the manufacturing labor cost element (and only the manufacturing labor cost element). However, a lack of quantitative responses prevented us from establishing a relationship between slope and material type/time frame. Nevertheless, although unable to quantify the effect, several respondents believed composites would have steeper improvement curve slopes than metals because many of the composite fabrication processes were dominated by manual operations and were new and therefore continually being improved. Another respondent suggested that slopes were more sensitive to automation levels than material type.

[13]The labor rates represent industry averages and include all costs for direct labor, facilities, capital cost of money, general and administrative expenses, and other direct charges (computer time, support services, travel, overtime, fringe benefits premium, and factored labor). Profit is excluded.

Table 20

INDUSTRY AVERAGE FULLY BURDENED LABOR RATES
(In 1990 dollars per hour)

Engineering labor	80.80
Tooling labor	70.60
Manufacturing labor	66.10
Quality assurance labor	65.60

SOURCE: These rates are based on an informal survey of six major aircraft firms that requested their identities not be disclosed.

Table 21

LATE 1980s: RECURRING DOLLARS PER POUND (INDUSTRY AVERAGE)
(Cumulative average dollars for 100 units for 1000 lb of structure in 1990 dollars)

Cost Element	Al	Al-Li	Ti	Steel	Gr/E	Gr/BMI	Gr/TP
Manufacturing labor[a]	603	685	977	722	1,071	1,263	1,111
Raw material[b]	31	83	88	20	150	168	199
Total manufacturing cost	634	768	1,065	742	1,221	1,431	1,310
Support labor[c]	302	326	493	410	675	713	798
Total recurring cost	935	1,094	1,558	1,153	1,896	2,144	2,108

[a]Includes fabrication and assembly labor.
[b]Includes material burden of 15 percent.
[c]Includes sustaining engineering, sustaining tooling, and quality assurance.

Table 22

MID-1990s: RECURRING DOLLARS PER POUND (INDUSTRY AVERAGE)
(Cumulative average dollars for 100 units for 1000 lb of structure in 1990 dollars)

Cost Element	Al	Al-Li	Ti	Steel	Gr/E	Gr/BMI	Gr/TP
Manufacturing labor[a]	529	620	830	718	880	1,092	958
Raw material[b]	25	28	82	20	118	127	137
Total manufacturing cost	554	648	912	738	998	1,219	1,095
Support labor[c]	277	314	440	411	550	597	606
Total recurring cost	831	962	1,352	1,149	1,548	1,816	1,701

[a]Includes fabrication and assembly labor.
[b]Includes material burden of 15 percent.
[c]Includes sustaining engineering, sustaining tooling, and quality assurance.

- On average, total recurring costs for composites are about twice those of aluminum and steel and 20 to 30 percent more than those of titanium.
- Recurring costs for both metals and composites are projected to decrease between the late 1980s and the mid-1990s. However, the improvement in composite costs (approximately 20 percent) is projected to be roughly twice the improvement in metal costs.
- Raw material costs are a small proportion of total recurring costs, typically less than 10 percent of the total and sometimes as little as 2 percent.
- Support labor costs account for 30 to 40 percent of total recurring costs.

Table 23 presents the range in overall recurring dollars per pound using the cluster ranges provided in earlier tables. Two observations regarding these ranges stand out. First, composite costs are subject to greater uncertainty than metal costs; the upper and lower bounds for the metals vary by a factor of two, and those for composites vary by a factor of three. However, the larger range expressed for composites is consistent with the fact that, compared with metals, they are a newer technology with considerably less design and production experience. The second observation concerns the absolute magnitude of the ranges. As stated earlier, part of the range can be explained by corporate-specific factors such as cumulative experience with a given material type and the level of factory automation. However, we believe that most of the range can be explained by part complexity. In general, for a given manufacturing technique, simple parts (large flat surfaces and nonload-bearing structure) will cost less per pound than complex parts (surfaces with compound curves, sine-wave spars, and load-bearing structure). Additionally, simple parts lend themselves to labor-saving automation in both the fabrication and inspection processes, whereas it is considerably more difficult to automate these processes for complex parts.

Part complexity can be reflected in either of two ways:

- *Type of aircraft.* Generally speaking, combat aircraft, because of their smaller size (which reduces the probability that automation will be cost-effective) and greater emphasis on performance (for example, higher speeds and greater maneuverability requirements, leading to more complex contoured structural elements) have greater part complexity than noncombat aircraft.
- *Level of material usage.* Generally speaking, as the level of usage of a given material type in a specific application increases, the part complexity also increases. This point is illustrated by the example provided in Table 24, which shows where composites would probably be incorporated into the structure of a generic fighter as the level of

Table 23

OVERALL RECURRING DOLLARS PER POUND (INDUSTRY MIN/MAX)
(Cumulative average dollars for 100 units
for 1000 lb of structure in 1990 dollars)

Cost Element	Al	Al-Li	Ti	Steel	Gr/E	Gr/BMI	Gr/TP
Late 1980s							
Minimum recurring	614	740	774	699	970	955	1,048
Maximum recurring	1,202	1,349	1,826	1,657	2,632	2,972	3,123
Mid-1990s							
Minimum recurring	521	612	703	691	822	812	869
Maximum recurring	1,045	1,207	1,649	1,511	2,133	2,499	2,597

Table 24

PART SELECTION AS A FUNCTION OF GRAPHITE COMPOSITION FOR A GENERIC FIGHTER

	Graphite Percent of Structure				
Component Type	10	25	35	45	55
Nonstructural access doors/panels	X	X	X	X	X
Structural access doors/panels	X	X	X	X	X
Vertical stabilizer skins	X	X	X	X	X
Horizontal stabilizer skins	X	X	X	X	X
Wing skins	X	X	X	X	X
Control surfaces	X	X	X	X	X
Speed brake	X	X	X	X	X
Landing gear doors	X	X	X	X	X
Additional doors/panels		X	X	X	X
Additional control surfaces		X	X	X	X
Spars, ribs		X	X	X	X
Shear webs, skin panels			X	X	X
Longerons				X	X
Frames, formers				X	X
Bulkheads					X

SOURCE: Aircraft contractor.

usage rises. It clearly demonstrates that as the level of substitution of composites for metals rises, the complexity of the application also rises.

SUMMARY

Industry responses to a data collection worksheet have been summarized in this section. The average, minimum, and maximum values for nonrecurring and recurring costs have been reported by material type. Responses show that costs do vary by material type and that on a per pound basis composite materials are more expensive than metal materials. Weight savings can mitigate the additional recurring costs. However, the values reported indicate it is unlikely the savings will be great enough to offset the additional costs.

V. SUGGESTED METHODOLOGY FOR ASSESSING THE EFFECT OF STRUCTURAL MATERIAL COMPOSITION ON OVERALL AIRFRAME COST

This section suggests an approach for aggregating the previously developed material cost factors into an overall structure cost. However, structure-related costs are only a part of overall airframe cost and are not normally considered in isolation.[1] As indicated in Table 25, *on average*, structure "accounts" for less than half of engineering (nonrecurring and recurring) and for 60 to 70 percent of manufacturing labor, manufacturing material, and quality assurance. Consequently, the approach that is outlined is one that considers overall airframe cost—that is, one that takes into account not only the cost of the airframe structure but also the cost of the airframe subsystems and final assembly/integration.

The basic inputs that the user *must* provide to exercise the methodology are as follows:

- Aircraft empty weight (lb).
- Maximum speed (kn).
- Number of flight test aircraft. (Input list continues on next page.)

Table 25

PERCENTAGES OF FUNCTIONAL COST ELEMENTS ATTRIBUTABLE TO STRUCTURE

Aircraft	Nonrecurring Engineering	Nonrecurring Tooling	Recurring Engineering	Recurring Tooling	Recurring Manufacturing Labor	Recurring Manufacturing Material	Recurring Quality Assurance
A-6	47	80	47	80	33	—	—
A-7	20	94	16	92	81	83	—
A-10	45	88	20	24	53	—	—
B-52	24	88	23	92	88	—	—
C-5	63	93	63	93	75	57	81
C-130	61	80	61	80	83	83	75
C-141	53	80	53	80	73	63	59
F-4	51	94	51	94	50	69	—
F-14	58	94	50	94	66	35	60
F-15	43	87	43	87	74	—	—
F-16	12	68	19	69	45	16	—
KC-135	58	100	57	100	88	—	—
Average	45	87	42	82	67	58	69

SOURCE: Proprietary RAND Aircraft Airframe Group Manhour and Cost Database. Dashes indicate missing or incomplete data.

[1]The term *airframe cost* refers to the cost of the assembled structural and aerodynamic components of the air vehicle that support subsystems essential to a particular mission. It includes not only the basic structure (wing, fuselage, empennage, and nacelles), but also the air induction system, starters, exhausts, fuel control system, inlet control system, alighting gear (tires, tubes, wheels, brakes, hydraulics, etc.), secondary power, furnishings (cargo, passenger, troop, etc.), engine controls, instruments (flight navigation, engine, etc.), environmental control, racks, mounts, intersystem cables and distribution boxes, etc., inherent to and inseparable from the assembled structure, dynamic systems, and other equipment homogeneous to the airframe. Airframe costs also encompass the integration and installation of the propulsion, avionics, and armament subsystems into the airframe but not those efforts directly related to their development and manufacture. (For additional clarification, see Department of Defense, MIL-STD-881, *Work Breakdown Structures for Defense Material Items*, 25 April 1975.)

- Type of aircraft (cargo or noncargo).
- Total structure weight[2] by material type.
- Percentages of functional cost elements attributable to structure.

Default values are provided for all other necessary inputs.

DESCRIPTION OF APPROACH

Described below is a method for estimating the development and production costs of aircraft airframes that is: sensitive to the material composition of the structure and *suitable for use in a program's conceptual stage when little detailed information is available.* Reduced to its simplest form, the method applies weighted material indexes to baseline CERs assumed to be representative of all-aluminum aircraft. It provides separate CERs and material indexes for the following major cost elements:[3]

- Nonrecurring engineering.
- Nonrecurring tooling.
- Development support.
- Flight test.
- Recurring engineering.
- Recurring tooling.
- Recurring manufacturing labor.
- Recurring manufacturing material.
- Recurring quality assurance.

The method does not, however, address the following:

- Costs of engines, avionics equipment, and armament.
- Training, support equipment, data, and spares.

Baseline CERs

The baseline CERs are listed in Table 26 and are identical to those documented in R-3255-AF[4] with the following exceptions:

- The engineering and tooling equations have been split into separate nonrecurring and recurring components.
- The manufacturing material, development support, and flight test CERs have been updated from FY77 dollars to FY90 dollars.

They were derived from a database consisting of 13 military aircraft with first flight dates ranging from 1960 to 1978: A-6, A-7, A-10, C-5, C-141, F-4, F-14, F-15, F-16, F-18, F-111, S-3A, and T-39. Empty weights for the sample aircraft range from under 10,000 lb to over 300,000 lb, while speeds range from 400 kn to over 1,300 kn.

[2]For purposes of this methodology, *structure weight* includes the weight of the wing, fuselage, empennage, nacelle, and air induction groups. It does not include the weight of the alighting gear group. (For additional clarification, see DoD, MIL-STD-1374A, *Weight and Balance Data Reporting Forms for Aircraft.*)

[3]Cost element definitions are provided in App. C.

[4] Hess and Romanoff, 1987.

Table 26

AIRFRAME CERS BASED ON SAMPLE OF 13 POST-1960 AIRCRAFT

Equation	R^2	SEE	F	N
$NRENGR = .0168\ EW^{.747}\ SP^{.800}$ (.000) (.016)	.76	.46	16	13
$NRTOOL = .0186\ EW^{.810}\ SP^{.579}$ (.000) (.003)	.92	.25	61	13
$DS = .0563\ EW^{.630}\ SP^{1.30}$ (.016) (.012)	.54	.82	6	13
$FT = 1.54\ EW^{.325}\ SP^{.822}\ TESTAC^{1.21}$ (.032) (.037) (.010)	.83	.48	15	13
$ENGR_{100} = .000306\ EW^{.880}\ SP^{1.12}$ (.008) (.058)	.58	.87	7	13
$TOOL_{100} = .00787\ EW^{.707}\ SP^{.813}$ (.000) (.007)	.80	.40	20	13
$LABR_{100} = .141\ EW^{.820}\ SP^{.484}$ (.000) (.013)	.88	.31	38	13
$MATL_{100} = .540\ EW^{.921}\ SP^{.621}$ (.000) (.003)	.91	.30	51	13
$QA_{100} = .076\ LABR_{100}$ (cargo)	—	—	—	2
$= .133\ LABR_{100}$ (noncargo)	—	—	—	11

R^2 = coefficient of determination; SEE = standard error of estimate (logarithm); F = F-statistic; N = sample size. Numbers in parentheses are significance levels of individual variables.

DS = Development support cost (thousands of 1990 dollars)

$ENGR_{100}$ = Cumulative recurring engineering hours for 100 aircraft (thousands)

EW = Aircraft empty weight (lb)

FT = Flight test cost (thousands of 1990 dollars)

$LABR_{100}$ = Cumulative recurring manufacturing labor hours for 100 aircraft (thousands)

$MATL_{100}$ = Cumulative recurring manufacturing material cost for 100 aircraft (thousands of 1990 dollars)

NRENGR = Nonrecurring engineering hours (thousands)

NRTOOL = Nonrecurring tooling hours (thousands)

QA_{100} = Cumulative recurring quality assurance hours for 100 aircraft (thousands)

SP = Maximum speed (kn)

TESTAC = Number of flight-test aircraft

$TOOL_{100}$ = Cumulative recurring tooling hours for 100 aircraft (thousands)

Application of weighted material indexes to these CERs requires that the following key assumptions be made:

- That the CERs are representative of the baseline material (aluminum).
- That the CERs are representative of late 1980s manufacturing technology.

Both assumptions are admittedly heroic. For example, as shown in Table 27, the aluminum content of at least three of the fighters in the database is only 50 percent. And substantial improvements in manufacturing technology have been made over the last three decades.

Table 27

AIRFRAME MATERIALS UTILIZATION
Percentage of Airframe Structure Weight

	Aircraft (first flight date)					
Material	F-4 (1961)	F-111 (1967)	F-14 (1970)	F-15 (1972)	F-16 (1976)	F-18 (1978)
Aluminum	70	59	48	52	79	48
Titanium	9	5	29	40	2	14
Steel	16	33	22	5	4	15
Composites		1	1	2	5	11
Other	5	2		1	10	12

For example, if the F-4 had been produced using F-15 manufacturing techniques, the unit cost of the 155th aircraft would have been 12–1/2 percent lower than it actually was.[5] Unfortunately, there is not much that can be done to alleviate these difficulties.[6] To the extent that the CERs actually embody more advanced material types and older manufacturing techniques than we have claimed, the bias is clear: Applying late 1980s material cost factors to the CERs will result in somewhat higher estimates than we would otherwise have obtained.[7]

Weighted Material Indexes

The weighted material indexes applied to each cost element are calculated as follows:

$$WMCF_j = STRFRAC_j \times \left[\sum RMCF_{i,j} \times (STRWT_i/\sum STRWT_i)\right] + \left[1 - STRFRAC_j\right]$$

where $WMCF_j$ = Weighted material cost factor for cost element 'j'
$STRFRAC_j$ = Portion of cost element 'j' attributable to structure (Table 25)
$RMCF_{i,j}$ = Material 'i' structural cost index for cost element 'j'
$STRWT_i$ = Material 'i' structural weight

The relative material cost factors (RMCFs) are provided in Table 28. They have been calculated on the basis of the *average* values presented in Sec. IV. For each cost element, the late-1980s aluminum value was considered the baseline. The complexity factors represent the ratios of cumulative average values for 100 airframes normalized to a weight of 1000 lb. Thus,

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[5]Stekler, 1985, p. 426.

[6]One possibility would be to try to incorporate a material index directly into the CERs as part of the regression analysis. However, this is a highly uncertain proposition since it would require the development of *historical* material cost factors, an effort that we believe would have little chance of success.

[7]Another concern with respect to the applicability of the CERs to future aircraft has to do with the nonstructural airframe subsystems (e.g., electrical, hydraulics, environmental control, and fuel system). Implicitly, we assume that there will be no change. However, as lower radar cross-section and greater maneuverability are sought, many of the nonstructural subsystems will be pushed to their limits. While an examination of such effects was beyond the scope of this study, we believe the direction of change is unambiguous: Subsystem costs will increase relative to the database used to derive our CERs.

Table 28

RELATIVE MATERIAL COMPLEXITY FACTORS
(Late 1980s, aluminum - 1.00)

Time Period/ Material Type	Nonrecurring Engineering	Nonrecurring Tooling	Recurring Engineering	Recurring Tooling	Recurring Manufacturing Labor	Recurring Manufacturing Material	Recurring Quality Assurance
Late 1980s							
Aluminum	1.0	1.0	1.0	1.0	1.0	1.0	1.0
Al-lithium	1.1	1.2	1.1	1.1	1.1	2.7	1.1
Titanium	1.1	1.4	1.4	1.9	1.6	2.8	1.6
Steel	1.1	1.1	1.1	1.4	1.2	0.7	1.4
Graphite/epoxy	1.4	1.6	1.9	2.2	1.8	4.9	2.4
Graphite/bismaleimide	1.5	1.7	2.1	2.3	2.1	5.5	2.5
Graphite/thermoplastic	1.7	2.0	2.9	2.4	1.8	6.5	2.6
Mid-1990s							
Aluminum	1.0	1.0	0.9	0.9	0.9	0.8	0.9
Al-lithium	1.0	1.1	1.0	1.1	1.0	0.9	1.0
Titanium	1.0	1.4	1.2	1.6	1.4	2.7	1.4
Steel	1.1	1.1	1.1	1.4	1.2	0.7	1.4
Graphite/epoxy	1.2	1.4	1.5	2.0	1.5	3.8	1.8
Graphite/bismaleimide	1.3	1.5	1.6	2.1	1.8	4.1	2.1
Graphite/thermoplastic	1.4	1.6	1.4	2.4	1.6	4.4	2.0

to the extent the user believes these ratios vary with either the quantity or the weight, they should be adjusted accordingly.[8]

Steps in Applying the Methodology

Assess Applicability. The first step in applying this method is to ensure that the empty weight and speed of the proposed aircraft lie within the range of the estimating sample:

Aircraft empty weight (lb): 9,753 to 320,085
Speed (kn): 400 to 1300+

Additionally, the estimator should ensure that the proposed aircraft does not differ substantially from the estimating sample along such other key dimensions as:

- The nature and stringency of the design requirements (e.g., radar signature requirements may be such that the tolerances, contours, and surface coatings of the proposed aircraft do not reflect the estimating sample).
- The type of programmatic strategies to be employed (e.g., the classification of a proposed aircraft program as "special access" would clearly set it apart from the programs in the estimating sample).

Apply Basic CERs. Once applicability has been assessed, values for each of the CERs listed in Table 26 should be determined. For those CERs estimated in manhours, the following fully burdened labor rates (in FY90 dollars) can be used to convert hours to equivalent dollars:

Engineering	80.80
Tooling	70.60
Manufacturing	66.10
Quality assurance	65.60

Calculate Adjustment Factors and Apply to Basic Values. Finally, the weighted material cost factors should be calculated for each cost element. Default (average) values for the cost element structural fractions and relative material complexity factors are provided in Tables 25 and 28, respectively. However, if the estimator has information that he or she believes will more accurately characterize the proposed program, then that information should be used. For example, consider the wide variability in the nonrecurring engineering structural fractions presented in Table 25; values run from a low of 12 percent (F-16) to a high of 63 percent (C-5). In percentage terms, structure-related nonrecurring engineering was five times as important on the C-5 program as on the F-16 program. However, we know the F-16 development program was preceded by fairly "hi-fidelity" prototypes, whereas the C-5 program was a very challenging, completely new development without prototypes. Moreover, the F-16 incorporated an engine that was already in service (the F100) while the C-5 used the TF-39, which was not only new but the highest thrust engine in the world at the time. Many such observations can be made regarding the values in Table 25. The estimator should not disregard such observations when deciding on appropriate structural fractions.[9]

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[8]However, we found no evidence in the industry survey responses to suggest that these *ratios* do vary with quantity or weight.

[9]Similar arguments can be made with respect to the relative material complexity factors presented in Table 28. For example, these ratios might be adjusted upward if the estimator believes that a company's experience with an advanced material type is less than the industry average.

Adjusting Values to Quantities Other Than 100. To adjust recurring values to quantities other than 100, improvement curve slopes are suggested as shown in Table 29.

APPLICATION TO HYPOTHETICAL AIRCRAFT

The just-described approach is now applied to two hypothetical aircraft. As indicated in Table 30, both planes have been defined such that their weight and speed are within the limits of the estimating sample and both have been assumed to be conventional acquisitions with respect to design and programmatic considerations. The only differences between the two aircraft are in the weights. The structure of Aircraft 1 is all aluminum and weighs 13,000 lb, while the structure of Aircraft 2 is 55 percent aluminum and 45 percent graphite and weighs only 11,700 lb. We assumed that only 5,200 lb of graphite/epoxy would be required to replace 6,500 lb of aluminum, a 20 percent weight savings.[10]

Table 29

FUNCTIONAL COST ELEMENT IMPROVEMENT CURVE SLOPES

Cost Element	Cumulative-Total Slope (%)[a]	Exponent
Recurring engineering	140	.485
Recurring tooling	146	.546
Manufacturing labor	156	.641
Manufacturing material	174	.799
Quality assurance	156	.641

[a]Equivalent cumulative-average slopes may be found by dividing the cumulative-total slopes by two.

Table 30

REQUIRED INPUTS

Input	Case 1: All-Aluminum Structure	Case 2: 55% Aluminum/ 45% Graphite/epoxy Structure
Empty weight (lb)	27,000	25,700
Speed (kn)	1,300	1,300
Number of test aircraft	20	20
Structure weight (lb)		
Aluminum	13,000	6,500
Graphite/epoxy	0	5,200
Aircraft type	Fighter	Fighter
Cost factor time frame	Late 1980s	Late 1980s
Unusual design requirements	None	None
Unusual programmatic considerations	None	None

[10]Potentially, even greater weight savings than suggested here are possible through the process of aircraft "resizing." As discussed in Sec. II, the potential for resizing depends on the aircraft requirements and the "freedom" to make

The results are presented in Table 31. The most important observation with respect to these cases is that based on today's manufacturing technology, the substitution of graphite/epoxy for half the aluminum in the structure will increase nonrecurring costs by about 4 percent and recurring costs by roughly 35 percent.[11]

Finally, the effect of projected manufacturing improvements on airframe cost is shown in Table 32. As indicated, relative to the late 1980s, nonrecurring costs in the mid-1990s are projected to decrease about 3 percent while recurring costs are projected to drop about 9 percent.

Table 31

EFFECT OF COMPOSITE SUBSTITUTION ON AIRFRAME COST
(Millions of FY90 dollars)

Cost Element	Case 1: All-Aluminum Structure	Case 2: 55% Aluminum/ 45% Graphite/epoxy Structure
Nonrecurring		
Engineering	859	895
Tooling	324	385
Development support	389	378
Flight test	578	568
Total nonrecurring	2150	2226
Recurring (100 airframes)		
Engineering	603	675
Tooling	257	358
Labor	1289	1537
Material	559	1078
Quality assurance	170	234
Total recurring	2878	3882
Total	5028	6108

design decisions (particularly with respect to the engine). As a result, the resizing of an aircraft is a complex process that does not lend itself to simple rules of thumb. Since our example is strictly illustrative, we have chosen not to address this aspect. However, in conducting real-world tradeoffs, the cost analyst should consult a qualified engineer regarding the resizing issue.

[11]This conclusion assumes that we started with an all-aluminum aircraft and that the only weight saving resulted directly from the material substitution. Different material mixes as well as consideration of aircraft resizing effects would produce different results.

Table 32

EFFECT OF PROJECTED MANUFACTURING IMPROVEMENTS
ON AIRFRAME COST
(millions of FY90 dollars)

Cost Element	Case 2: 55% Aluminum/ 45% Graphite/epoxy Structure; Late 1980s	Case 3: 55% Aluminum/ 45% Graphite/epoxy Structure; Mid-1990s
Nonrecurring		
Engineering	895	862
Tooling	385	360
Development support	378	378
Flight test	568	568
Total nonrecurring	2226	2168
Recurring (100 airframes)		
Engineering	675	619
Tooling	358	328
Labor	1537	1379
Material	1078	890
Quality assurance	234	198
Total recurring	3882	3414
Total	6108	5582

VI. CONCLUSIONS

This report identifies, describes, and quantifies the cost effects of structural materials likely to be incorporated into aircraft becoming operational in the 1990s. More specifically, it summarizes cost information obtained from aircraft prime contractors for the materials, functional cost elements, and time periods given in Table 33.

RELATIVE MATERIAL COSTS

The data indicate that both nonrecurring and recurring costs per pound are higher for composites than for metals. In most cases, these costs are considerably higher. Nonrecurring engineering for composite structure is between 40 and 70 percent higher than for aluminum structure, largely because of the "newness" of composites (properties are not standardized and analytic tools for design are still being developed). Similarly, nonrecurring tooling for composites is between 60 and 100 percent greater than for aluminum, largely because of the materials used to make the tools, the complexity of part shapes (e.g., compound curvatures), and the required part accuracy (tolerances).

On the production side, the overall recurring cost per pound[1] of composite is about twice that of aluminum and steel and 20 to 30 percent more than that of titanium. Average hours-per-pound factors for aluminum and graphite/epoxy (the most technologically mature of the composites) for the late 1980s time frame are as given in Table 34. With one exception, the graphite/epoxy factors are all roughly twice the aluminum values. Moreover, the effect of the single exception (raw material costs, which vary by a factor of almost 5) is substantially mitigated by the fact that material costs typically constitute less than 10 percent of total recurring costs.

The data also indicate that recurring costs for both metals and composites can be expected to decrease between the late 1980s and the mid-1990s. However, the improvement in

Table 33

MATERIALS, COSTS, AND TIME PERIODS

Material Types	Cost Elements	Time Periods
Metals	Nonrecurring	Late 1980s
Aluminum	Engineering	Mid-1990s
Aluminum-lithium	Tooling	
Titanium	Recurring	
Steel	Engineering	
Composites	Tooling	
Graphite/epoxy	Manufacturing labor	
Graphite/bismaleimide	Quality assurance	
Graphite/thermoplastic	Manufacturing material	

[1]Normalized for weight (1000 lb) and quantity (100).

Table 34

ALUMINUM AND GRAPHITE/EPOXY COST COMPARISON

Cost Element	Aluminum	Graphite/ Epoxy	Ratio
Recurring engineering (hr/lb)	1.0	1.9	1.9
Recurring tooling (hr/lb)	1.6	3.6	2.2
Recurring manufacturing labor (hr/lb)	9.1	16.2	1.8
Recurring quality assurance (hr/lb)	1.7	4.1	2.4
Recurring manufacturing material ($/lb)	27	130	4.8

composite costs (approximately 20 percent) is projected to be roughly twice that of metal costs, an outcome we believe is consistent with the "newness" of composites technology and its larger scope for improvement. However, these projections were developed during a more optimistic time period (1987). In the current environment, where programs are being stretched out or canceled, companies may be less willing (or able) to make the capital investments necessary to achieve these projections.

ASSESSING THE EFFECT OF STRUCTURAL MATERIAL COMPOSITION ON OVERALL AIRFRAME COST

Structure-related costs are only a part of overall airframe costs and are not normally considered in isolation. Consequently, we have outlined an approach that considers overall airframe cost—that is, a method that takes into account not only the cost of the airframe structure but also the cost of the airframe subsystems and final assembly/integration. It is suitable for use in a program's conceptual stage when little detailed information is available.

Reduced to its simplest form, the method applies weighted material indexes to baseline CERs that are assumed to be representative of all-aluminum aircraft. The CERs, which are virtually identical to those documented in R-3255-AF, were derived from a database consisting of 13 post-1960 military aircraft.[2] Empty weights for the sample aircraft range from under 10,000 lb to over 30,000 lb, while speeds range from 400 kn to over 1,300 kn. The recommended material indexes were calculated on the basis of the *average* values presented in Sec. IV. For each cost element, the late 1980s aluminum value was considered the baseline.

The method was applied to two hypothetical aircraft, one having an all-aluminum structure weighing 13,000 lb and the other having a structure that was 55 percent aluminum and 45 percent graphite/epoxy and weighing 11,700 lb.[3] The result, which is based on late 1980s manufacturing technology, shows that the substitution of graphite epoxy for half the aluminum in the structure will increase nonrecurring costs by about 3 percent and recurring costs by roughly 35 percent. Other material variations will produce different results.

[2]The A-6, A-7, A-10, C-5, C-141, F-4, F-14, F-15, F-16, F-18, F-111, S-3, and T-39.

[3]We assumed that only 5200 lb of graphite/epoxy would be required to replace 6500 lb of aluminum, a 20 percent weight savings.

FUTURE WORK

No study of the type described here is ever complete. However, at some point, it is necessary to call a halt, present the results, and move on to something else. Nevertheless, there are several avenues that might profitably be pursued in subsequent efforts.

Update of Material Cost Factors

As aircraft with higher percentages of composite usage begin to enter service, additional design and production cost data will be accumulated. Presumably, these additional data could help to reduce the large uncertainty now associated with the costing of composites. Additionally, other more advanced materials are currently under development for the National Aerospace Plane (NASP),[4] including carbon-carbon, titanium-aluminide, and silicon-carbide/titanium. Consequently, at some point in the future, it would undoubtedly be beneficial to update the factors presented herein as well as to collect cost data for the more technologically advanced NASP materials.

Consideration of O&S Factors

If weight is saved, composite aircraft will have lower fuel costs than their metal counterparts. However, to date, the military services have accumulated only very limited composite maintenance and repair experience. Moreover, that experience has generally been obtained on aircraft that use small amounts of composites in secondary structures. Future aircraft promise to use much greater amounts of composites and in more critical areas. Further complicating the issue are the trends toward unitized design (larger, more integral, parts) and stealth enhancement, including tighter tolerance requirements and the incorporation of radar-absorbing materials/structure. Unfortunately, repair concepts, inspection techniques, sparing philosophy, and required personnel skills for the support of sophisticated composite aircraft are not well defined. At some point, however, the costs associated with maintaining alternative structural materials will need to be addressed.

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[4]Temperatures on NASP's nose and leading edges will exceed 3000°F, while temperatures over most of the remaining surface structure will be on the order of 1500°F.

Appendix A

SAMPLE WORKSHEET

This appendix presents the worksheet upon which the industry survey was based. The five principal sections are:

I. Corporate material experience
II. Material usage (subassemblies where material used)
III. Material technical characteristics
IV. Cost data
V. General information

Nonproprietary qualitative data requested in worksheet sections I, II, III, and IV were incorporated into the narrative throughout the report. Simple averages (across companies) of the cost data are presented in Sec. IV.

I. CORPORATE MATERIAL EXPERIENCE

I. CORPORATE HISTORY

Please list both previous and planned experience with major subassemblies using composite materials or advanced metal alloys. (Corporate brochures welcome.)

PRODUCTION PROGRAMS

Aircraft*	Type of Subassembly	Weight (lb) per Shipset	Advanced Material Used	Quantity Produced	Year of First Application

RESEARCH AND DEVELOPMENT PROGRAMS

Aircraft*	Type of Subassembly	Weight (lb)	Advanced Material Used	Quantity (if applicable)	Year of Application

*If for security reasons the aircraft type, or any item, cannot be designated, we would still appreciate a response to the items that you can complete. It is not necessary to restrict responses to military aircraft programs. Any programs that may provide useful information for an aircraft program are appropriate.

II. MATERIAL USAGE
(subassemblies where material used)

MATERIAL NAME: ______________________________

Check the boxes that characterize the material's use in the subassemblies listed below. Place a "C" in the box if the material currently is being used in the subassembly. Place a "P" in the box if the material is projected to be used. Subassembly definitions are included in the glossary.

TYPICAL USES OF COMPOSITES/METAL ALLOYS WITHIN MAJOR AIRFRAME SUBASSEMBLIES						
	Nonstructural		Primary Structure		Secondary Structure	
Subassembly	Simple	Complex	Simple	Complex	Simple	Complex
Wing						
Fuselage — forward — mid — aft						
Empennage						
Nacelle						
Landing Gear						

Are there other factors you believe affect a part's complexity (e.g., shape, size, number of interfaces)?

____

____

____

____

____

____

III. MATERIAL TECHNICAL CHARACTERISTICS

Please complete the following questions for the production aircraft that provide the basis for the cost information on the subsequent pages.

MATERIAL NAME: ______________________________
(identify the series or designation)

MATERIAL TECHNICAL CHARACTERISTICS

Density (psi): ______________________________

Strength (psi × 10^6): ______________________________

Modulus (psi × 10^6): ______________________________

Strain (%): ______________________________

Maximum operating temperature (°F): ______________________________

Alloys—composition: ______________________________

RAW MATERIAL FORM: ☐ COMPOSITE: ____ Tape ____ Broadgood ____ Other (please specify) ________

☐ METAL/ALLOY: ____ Sheet ____ Plate ____ Billet ____ Forged ____ Extrusion

____ Powder ____ Other (please specify) ________

TYPE OF AIRCRAFT* ☐ Fighter/Attack ☐ Transport ☐ Bomber ☐ Helicopter

☐ Commercial ☐ General Aviation ☐ Other (please specify) ________

PERFORMANCE: ☐ Subsonic ☐ Supersonic

DESIGN CONCEPT: ☐ Traditional metal design ☐ Composite design ☐ Other (please specify) ________

LEVEL OF AUTOMATION IN MANUFACTURING: ☐ Manual ☐ Semi-Automated ☐ Fully Automated

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*Again, if the type of aircraft cannot be designated for security reasons, we still would appreciate responses to any questions that you can answer.

IV. COST DATA

MATERIAL NAME: ________________________________

COST INFORMATION

For the requested cost information, please assume a notional aircraft with the following material breakdown:

_______ % Material named above

_______ % Aluminum

_______ % Titanium

_______ % Other ________________________________

___________ Estimated airframe unit weight (lb)

ASSUMPTIONS

FY 87$ (as of 1 April 1987). Ratios relative to conventional aluminum airframe.

	Present (1987)		Projection (1995)	
	Low	High	Low	High
NONRECURRING RATIOS*				
Engineering —labor hours ratio				
Tooling —initial tooling labor hours ratio				
Investment —investment $**				
Principal reasons for differences between time frames (e.g., material supply and demand, level of factory automation, processing technology, design technology, etc.)				

*Ratios relative to a conventional aluminum airframe (e.g., Al = 1.00).

**The dollar value of new capital equipment needed for the advanced material (e.g., autoclaves, X-ray test equipment, filament winding machines, tape lay-up machines, etc.).

IV. COST DATA (continued)

MATERIAL NAME: ______________________

ASSUMPTIONS

Cumulative average costs for 100 units. FY 87$ (as of 1 April 1987).

	Present (1987)		Projection (1995)	
	Low	High	Low	High
RECURRING VALUES*				
Engineering —labor hours/lb				
Manufacturing —raw material $/lb				
—buy-to-fly ratio				
—weight sizing factor (%)**				
—labor hours				
—material				
—fabrication labor hours/lb				
—fabrication labor improvement curve slope (%)				
—assembly labor hours/lb				
—assembly labor improvement curve slope (%)				
—integration labor hours/lb				
—integration labor improvement curve slope (%)				
Sustaining tooling labor —hours/lb				
Quality assurance —labor hours/lb				
Principal reasons for differences between time frames (e.g., material supply and demand, level of factory automation, processing technology, design technology, etc.)				

*Estimated values for finished airframe. Please identify actual values with an "A" and estimated values with an "E" after the entry.

**ARCO factor.

V. GENERAL INFORMATION*

How do tooling requirements change with the advent of composite materials and new alloys? What are the factors in the decision to use metal tools versus composite tools?

Are there any new materials or manufacturing technologies not mentioned above that may substantially alter cost expectations in the future? Briefly describe them.

Are there components of total weapon life cycle cost that you believe may be substantially altered (up or down) by composite materials/new alloys (e.g., development, production, operations, and support costs)? How and why?

As a result of composite materials or new metal alloys do you believe there will be a shift in the typical funding profile for an aircraft program?

Additional comments.

*Please attach additional sheets if necessary.

Appendix B

OTHER STUDIES OF COMPOSITE MATERIAL COSTS

Because of the underlying differences in the design and manufacturing technology of composite materials and conventional metals, historical cost databases and models based on metals technology may not be appropriate indicators of the cost of future aircraft systems. We have identified four studies conducted in the 1980s that addressed the cost of composite materials in aircraft structure. They concentrated on recurring manufacturing costs at the total airframe level, probably because of the limitations of the data. Only one has attempted to go below this level. The studies are:

- Scott Woo and William Ditto, *Cost Estimating Relationships for Army Helicopter Composite Airframes,* Army Aviation Systems Command, 1983.
- Major Gordon Kage II *Estimating the Cost of Composite Material Airframes Using the RAND Corporation Development and Procurement Costs of Aircraft Parametric Model,* Air Force Institute of Technology, 1983.
- Walt Pingel, *Aerospace Industry Composites Survey,* Air Force Aeronautical Systems Division, 1986.
- Kirk Hoy et al., *Advanced Composites Airframe Cost Model,* MCR, Inc., 1987.

Generally speaking, the two studies performed in 1983 estimated that the substitution of composite materials for metals would *reduce* cost. The two later studies, however, concluded that the substitution of composite materials would increase costs. Material categorization also varies between these studies; some simply separate metals from composites, and others collect information by the specific material types.

ESTIMATING THE COST OF COMPOSITE MATERIAL AIRFRAMES USING THE RAND CORPORATION DEVELOPMENT AND PROCUREMENT COSTS OF AIRCRAFT PARAMETRIC MODEL (DAPCA III)

This study was conducted in 1983 by Major Gordon Kage II as an Air Force Institute of Technology master's thesis. His objective was to develop a series of indices that reflect the differences in manufacturing cost for composite materials and metals for use with the RAND Development and Procurement Cost of Aircraft Model version III. Indexes were developed for nonrecurring tooling hours, recurring manufacturing hours, and material dollars.

These indices were developed using the ICAM and FACET models. ICAM provided information for metal parts, and FACET provided information for composite parts. Both models require detailed information and estimate at the part level. Descriptive data for 30 detailed parts ranging from complex fuselage structure to simple spars and panels were collected. The descriptive data were input into the ICAM and FACET models and the resulting estimates were compared to develop the indices. The indices were all less than 1.0 when composites are incorporated, contrary to the data presented in the RAND, ASD, and MC&R reports.

COST ESTIMATING RELATIONSHIPS (CERS) FOR ARMY HELICOPTER COMPOSITE AIRFRAMES

This report was issued in 1983 by the U.S. Army Aviation Systems Command and was prepared by Scott Woo and William Ditto. A methodology was developed that uses indices to adjust a helicopter recurring CER for advanced technology in the 1990s. The indices were based on 1980 RAND data for V/STOL aircraft, and 1980 and 1983 Grumman data used in their Modular Life Cycle Cost Model (MLCCM). These indices are to be used in conjunction with a recurring manufacturing CER developed for the Army Time Phased Parametric Life Cycle Cost Model. Data are presented that cover historical helicopter recurring manufacturing costs and material distribution.

This report also presents data on historical helicopter recurring manufacturing costs derived from various program office baseline cost estimates, learning curve slopes for old and new technology helicopters, and airframe material composition. The helicopter data include: AH-1G, UH-1H, UH-60A, OH-60A, OH-58A, CH-47C, CH-54B, AH-64A, CH-47D, AHIP, and ACAP.

Woo and Ditto concluded that raw composite material costs were much higher than traditional metal material costs, and composite material labor costs were lower than metal labor costs. According to their calculations, overall composite structure is less expensive than metal structure because the weight savings and lower labor cost more than offset the higher material costs.

AEROSPACE INDUSTRY COMPOSITES SURVEY

This study was performed by Walt Pingel of the Air Force Aeronautical Systems Division in 1986. The two objectives of this study were to derive labor indices for selected composite materials and to establish general trends in composite material usage in the aerospace industry. Data were collected by means of a survey for manufacturing, tooling, and quality assurance, for each of 11 material types. The labor indices represent the ratio of the level of effort required for the advanced material relative to aluminum. These ratios represent a mixture of the nonrecurring and recurring portions of the cost elements. Industry average values are reported. Table B.1 summarizes the results. These factors must be used in conjunction with another estimating methodology, although no method was suggested.

ADVANCED COMPOSITES AIRFRAME COST MODEL

This study was prepared by Kirk Hoy et al., of MCR, Inc., and was originally conducted for the Navy in 1987 and subsequently updated for the Air Force in 1988. The objective of this study was to collect aircraft production data and develop CERs. Recurring manufacturing labor hours and material dollars for conventional metal and composite aircraft were collected. Aircraft included in the database are A-6E, A-7A/D, A-10A, AV-8B, F-4B, F-5E, F-14A, F-15A, F-16A, F/A-18, F-101C, F-111A, B-1B, the Boeing 737, and the Lockheed L-1011. Data were available at several levels for these programs. Most had total production hours available for a small subset of the work breakdown structure. A few had a breakdown of fabrication and assembly hours for particular elements in the work breakdown structure (WBS).

Cost estimating relationships were developed for airframe, fuselage, wing, wing/stabilizer, and empennage labor hours, and airframe material using regression analysis. Because of data

Table B.1

SUMMARY OF ASD MATERIALS SURVEY
(Industry averages of ratio of level of effort for advanced material relative to aluminum)

Material	Cost Element		
	Manufacturing	Tooling	Quality Assurance
Boron/epoxy	2.17	1.69	1.53
Boron/polyimide	2.41	1.81	1.59
Boron/aluminum	2.78	2.00	1.53
Kevlar	1.60	1.44	1.89
Carbon/carbon	3.71	2.42	1.78
Advanced aluminum	1.37	1.21	1.18
Graphite/epoxy	1.69	1.56	1.89
Graphite/aluminum	2.28	1.88	1.43
Graphite/nickel	2.53	2.00	1.43
Graphite/bismaleimide	1.92	1.69	1.46
Graphite/thermoplastic	2.00	1.75	1.86

limitations, several of these CERs were not recommended for use.[1] For each subassembly element the CERs took the linear form. Inputs to each relationship are metal weight and composite weight. As an example, the airframe estimating relationship is presented below:

Manufacturing labor hours for unit 100 =
$$6480.2 + 5.01 \times \text{(metal weight)} + 14.46 \times \text{(composite weight)}$$

Some information on support labor costs[2] and learning curve slopes is also presented.

[1]In particular, the wing, empennage, and airframe material relationships.
[2]Sustaining engineering, sustaining tooling, and quality assurance.

Appendix C

COST ELEMENT DEFINITIONS

Work breakdown structure categories included in the RAND airframe cost structure are shown in Table C.1.[1] A matrix that maps contractor cost-data reporting (CCDR) categories[2] and relevant WBS categories into specific RAND airframe cost elements is provided in Table C.2.

NONRECURRING ENGINEERING

In general, the nonrecurring engineering cost element encompasses the hours expended in the study, analysis, design, development, and evaluation of the basic airframe. More specifically, it includes engineering for design studies and integration; for wind-tunnel models, drop model, mockups, and propulsion-system tests; for laboratory testing of components, subsystems, and static and fatigue articles; for preparation and maintenance of drawings and process and materials specifications; and for reliability. Engineering hours not directly attributable to the airframe itself (those charged to ground-handling equipment, spares, and training equipment) are not included. Engineering hours expended as part of the tool and production-planning function are included with the cost element tooling; those expended as part of the flight-test planning and evaluation effort are included in the flight-test cost element. Material, purchased parts, and test equipment required to accomplish the engineering function are assumed to be included in the fully burdened engineering labor rate.

RECURRING ENGINEERING

Recurring engineering covers such things as customer support/liaison, identifying ways to correct operationally revealed deficiencies, and suggesting possible system improvements. Although the break between nonrecurring and recurring engineering is subjective, the point of segregation is frequently at some specified point such as "design freeze" or "90 percent engineering drawing release."

NONRECURRING TOOLING

Nonrecurring tooling encompasses the cost of the initial set of tools and all duplicate tools produced to permit a specific rate of production. In general, tooling refers only to those tools designed for use on a specific program—i.e., assembly tools, dies, jigs, fixtures, master forms, gauges, handling equipment, load bars, work platforms, and test and checkout equipment. General purpose tools such as milling machines, presses, routers, and lathes (except for the cutting instruments) are considered capital equipment. If such equipment is owned by the contractor (much of it is government-owned), an allowance for depreciation is included in the

[1]See MIL-STD-881, *Work Breakdown Structure for Defense Materiel Items.*

[2]See AFLCP 800-15, *Contractor Cost Data Reporting System.*

Table C.1

WBS CATEGORIES INCLUDED IN RAND AIRFRAME COST ELEMENTS

WBS Category	Included?
Air vehicle	
Airframe[a]	Yes
Propulsion[b]	No
Avionics[b]	No
Armament/weapon delivery[b]	No
Training	No
Peculiar support equipment	No
Systems test and evaluation	
Development test	Yes
Technical evaluation	Yes
Operational evaluation	Yes
Mockups	Yes
Test facilities	Yes
Other ST&E	Yes
System/project management	Yes
Data	
Engineering/management data	Yes
ILS data (tech orders and manuals)	No
Operational/site activation	No
Common support equipment	No
Industrial facilities	No
Initial spares and repair parts	No

[a]The term *airframe* refers to the assembled structural and aerodynamic components of the air vehicle that support subsystems essential to a particular mission. It includes, for example, the basic structure (wing, empennage, fuselage, and associated manual flight control system), the air induction system, starters, exhausts, the fuel control system, inlet control system, alighting gear (tires, tubes, wheels, brakes, hydraulics, etc.), secondary power, furnishings (cargo, passenger, troop, etc.), engine controls, instruments (flight navigation, engine, etc.), environmental control, racks, mounts, and intersystem cables and distribution boxes, etc., that are inherent to and inseparable from the assembled structure, dynamic systems, and other equipment homogeneous to the airframe. All efforts directly related to propulsion, avionics, and armament are excluded.

[b]Installation of the propulsion, avionics, and armament subsystems is accounted for in the airframe category; the design integration effort is included in system/project management; and some testing is included in system test and evaluation.

overhead account. Tooling hours include all effort expended in tool and production planning, design, fabrication, assembly, and installation as well as the programming and preparation of tapes for numerically controlled machines. The cost of the material used in the manufacture of the dies, jigs, fixtures, etc. is assumed to be included in the fully burdened tooling labor rate.

Table C.2

RAND AIRFRAME COST ELEMENTS

	CCDR Categories[a]									
	Engineering		Tooling		Manufacturing Labor		Manufacturing Material		Quality Control	
WBS Category	NR	R	NR	R	NR	R	NR	R	NR	R
Air vehicle										
Airframe	NEngr	REngr	NTool	RTool	DS	Labr	DS	Matl	QC	QC
Systems test and evaluation										
Development tests	NEngr	—	DS	—	DS	—	DS	—	DS	—
Technical evaluation	FT	—	FT	—	FT	—	FT	—	FT	—
Operational evaluation	FT	—	FT	—	FT	—	FT	—	FT	—
Mockups	NEngr	—	DS	—	DS	—	DS	—	DS	—
Test facilities	NEngr	—	DS	—	DS	—	DS	—	DS	—
Other ST&E	NEngr	—	DS	—	DS	—	DS	—	DS	—
System/project management	NEngr	REngr	DS	RTool	DS	Labr	DS	Matl	DS	QC
Engineering/management data	NEngr	REngr	—	—	—	—	—	—	—	—

[a]NR - nonrecurring; R - recurring; NEngr - nonrecurring engineering; REngr - recurring engineering; NTool - nonrecurring tooling; RTool - recurring tooling; Labr - manufacturing labor; Matl - manufacturing material; DS - development support; FT - flight test; QC - quality control.

RECURRING TOOLING

Recurring tooling includes tool maintenance, modification, rework, and replacement.

MANUFACTURING LABOR

Manufacturing labor is all the direct labor necessary to machine, process, fabricate, and assemble the major structure of an aircraft and to install purchased parts and equipment, engines, avionics, and ordnance items, whether contractor-furnished or government-furnished. Manufacturing manhours include the labor component of off-site manufactured assemblies and effort on those parts that because of their configuration or weight characteristics are design-controlled for the basic aircraft. These parts normally represent considerable proportions of airframe weight and of the manufacturing effort and are included regardless of their method of acquisition. Such parts specifically include actuating hydraulic cylinders, radomes, canopies, ducts, passenger and crew seats, and fixed external tanks. Manhours required to fabricate purchased parts and materials are excluded from the cost element. Nonrecurring labor undertaken in support of engineering during the development phase is included in the development support cost element.

MANUFACTURING MATERIAL

Manufacturing material includes raw and semifabricated materials plus purchased parts (standard hardware items such as electrical fittings, valves, and hydraulic fixtures) used in the manufacture of the airframe. This category also includes purchased equipment (i.e., motors, generators, batteries, landing gear, air conditioning equipment, instruments, and hydraulic and

pneumatic pumps), whether procured by the contractor or furnished by the government. Where such equipment is designed specifically for a particular aircraft, it is considered as subcontracted, not as purchased equipment, and is therefore included in the manufacturing labor cost element. Nonrecurring material used in support of engineering during the development phase is included in the development-support cost element.

DEVELOPMENT SUPPORT

Development support is the nonrecurring manufacturing effort undertaken in support of engineering during the development phase of an aircraft program. It is intended to include the manhours and material required to produce mockups, models, test parts, static and fatigue test items, and other hardware items (excluding complete flight-test aircraft) needed for airframe development.

FLIGHT TEST

Flight test includes all costs incurred by the contractor in the conduct of flight testing except production of the test aircraft. Engineering planning, data reduction, manufacturing support, instrumentation, all other materials, fuel and oil, pilot's pay, facilities, rental, and insurance costs are included. Flight-test costs incurred by the Air Force, Army, or Navy are excluded.

QUALITY CONTROL

Quality control refers to the hours expended to ensure that prescribed standards are met. It includes such tasks as receiving inspection; in-process and final inspection of tools, parts, subassemblies, and complete assemblies; and reliability testing and failure-report reviewing. The preparation of reports relating to these tasks is considered direct quality-control effort.

Appendix D

DEFINITION OF TERMS

The following definitions have been taken from *New Structural Materials Technologies: Opportunities for the Use of Advanced Ceramics and Composites,* QTA-TM-E-32, September 1986; and *DOD/NASA Advanced Composites Design Guide,* Volumes 1-4, Flight Dynamics Laboratory, Air Force Wright Aeronautical Laboratories, 1983.

adhesive	A substance capable of holding two materials together by surface attachment. In the Guide, the term is used specifically to designate structural adhesives, which produce attachments capable of transmitting substantial structural loads.
advanced filaments	Continuous filaments made from high strength, high modulus materials for use as a constituent of advanced composites.
alloy	A material having metallic properties and consisting of two or more elements.
anelasticity	A characteristic exhibited by certain materials in which strain is a function of both stress and time, such that while no permanent deformations are involved, a finite time is required to establish equilibrium between stress and strain in both the loading and unloading directions.
anisotropic	Showing different physical or mechanical properties in different directions.
aramid	Lightweight polyaromatic amide fibers having excellent high temperature, flame resistance, and electrical properties. These fibers are used as high strength reinforcement in composites.
aspect ratio	In an essentially two-dimensional rectangular structure (e.g., a panel), the ratio of the long dimension to the short dimension. However, in compression loading, it is sometimes considered to be the ratio of the load direction dimension to the transverse dimension. Also, in fiber micro-mechanics, it is referred to as the ratio of length to diameter.
autoclave	A closed vessel for producing an environment of fluid pressure, with or without heat, to an enclosed object while undergoing a chemical reaction or other operation.
autoclave molding	A process similar to the pressure bag technique. The layup is covered by a pressure bag, and the entire assembly is placed in an autoclave capable of providing heat and pressure for curing the part. The pressure bag is normally vented to the outside.
brittle fracture	A break in a brittle material due to the propagation of cracks originating at flaws.
carbon/graphite	These fibers, which are the dominant reinforcement in "advanced" composites, are produced by pyrolysis of an organic precursor—e.g.,

	polyacrylonitrile (PAN), or petroleum pitch—in an inert atmosphere. Depending on the process temperature, fibers having high strength or high elastic modulus may be produced.
ceramic	An inorganic, nonmetallic solid.
chemically bonded ceramics	Used here to distinguish advanced cements and concretes, which are consolidated through chemical reactions at ambient temperatures (generally involving uptake of water), from high-performance ceramics, such as silicon nitride and silicon carbide, which are densified at high temperatures.
compressive stress	A stress that causes an elastic body to shorten in the direction of the applied force.
consolidation of parts	Integration of formerly discrete parts into a single part that encompasses several functions, a key advantage of engineered materials such as ceramics and composites.
continuous fiber	A reinforcing fiber in a composite that has a length comparable to the dimensions of the structure.
continuous filament yarn	Yarn formed by twisting two or more continuous filaments into a single, continuous strand.
coupling agent	Any chemical substance designed to react with both the reinforcement and matrix phases of a composite material to form or promote a stronger bond at the interface. Coupling agents are applied to the reinforcement phase from aqueous or organic solution, from the gas phase, or added to the matrix as an integral blend.
crazing	The development of a multitude of very fine cracks in the matrix material.
creep	A time-dependent strain of a solid, caused by stress.
cross-linking	The formation of chemical bonds between formerly separate polymer chains.
crossply	Any filamentary laminate that is not uniaxial. Same as angleply. In some references, the term crossply is used to designate only those laminates in which the laminae are at right angles to one another, while the term angleply is used for all others. In the Guide, the two terms are used synonymously. With the advent of the Guide laminate orientation code, the reservation of a separate terminology for only one of several basic orientations is unwarranted.
crystal	A homogeneous solid in which the atoms or molecules are arranged in a regularly repeating pattern.
cure	To change the properties of a thermosetting resin irreversibly by chemical reaction—i.e., condensation, ring closure, or addition. Cure may be accomplished by addition of curing (cross-linking) agents, with or without catalyst, and with or without heat.
cure stress	A residual internal stress produced during the curing cycle of composite structures. Normally, these stresses originate when different components of a wet layup have different thermal coefficients of expansion.

debond	A deliberate separation of a bonded joint or interface, usually for repair or rework purposes.
deformation, plastic	
deformation	Any alteration of shape or dimensions of a body caused by stresses, thermal expansion or contraction, chemical or metallurgical transformations, or shrinkage and expansions due to moisture change.
delamination	The separation of the layers of material in a laminate. This may be local or may cover a large area of the laminate. It may occur at any time in the cure or subsequent life of the laminate and may arise from a wide variety of causes.
dielectric	A material that is an electrical insulator or in which an electric field can be sustained with a minimum dissipation of power.
disbond	An area within a bonded interface between two adherends in which an adhesion failure or separation has occurred. It may occur at any time during the life of the structure and may arise from a wide variety of causes. Also, colloquially, an area of separation between two laminae in a finished laminate (where "delamination" is preferred).
drapability	The ease in which a material may be formed into a complex contoured shape without undesirable features (i.e. folds, wrinkles, etc.).
ductility	The ability of a material to be plastically deformed by elongation without fracture.
E-glass	A borosilicate glass most used for glass fibers in reinforced plastics.
elasticity	The property whereby a solid material deforms under stress but recovers its original configuration when the stress is removed.
end	An individual warp yarn, thread, monofilament, or roving.
extrusion	A process in which a hot or cold semisoft solid material, such as metal or plastic, is forced through the orifice of a die to produce a continuously formed piece in the shape of the desired product.
fabric	A generic material construction consisting of interlaced yarns or fibers, usually a planar structure. Specifically, a cloth woven in an established weave pattern from advanced fiber yarns and used as the fibrous constituent in an advanced composite lamina. In a fabric lamina, the warp direction is considered the longitudinal (L) direction, analogous to the filament direction in a filamentary lamina.
failure	Collapse, breakage, or bending of a structure or structural element such that it can no longer fulfill its purpose.
fatigue	Failure of a material by cracking resulting from repeated or cyclic stress.
fiber	A single homogeneous strand of material, essentially one-dimensional in the macro-behavior sense, used as a principal constituent in advanced composites because of its high axial strength and modulus.
fiber content	The amount of fiber present in a composite, usually expressed as a percentage volume fraction or weight fraction of the composite.
fiber count	The number of fibers per unit width of ply present in a specified section of a composite.

fiber direction	The orientation or alignment of the longitudinal axis of the fiber with respect to a stated reference axis.
fiber system	The type and arrangement of fibrous material that constitutes the fiber constituent of an advanced composite. Examples of fiber systems are collimated filaments or filament yarns, woven fabric, randomly oriented short-fiber ribbons, random fiber mats, whiskers, etc.
filament	A variety of fibers characterized by extreme length, such that there are normally no filament ends within a part except at geometric discontinuities. Filaments are used in filamentary composites and are also used in filament winding processes, which require long continuous strands.
filamentary composites	A major form of advanced composites in which the fiber constituent consists of continuous filaments. Specifically, a filamentary composite is a laminate composed of several laminae, each of which consists of a nonwoven, parallel, uniaxial, planar array of filaments (or filament yarns) imbedded in the selected matrix material. Individual laminae are directionally oriented and combined into specific multiaxial laminates for application to specific envelopes of strength and stiffness requirements.
filament winding	An automated process in which continuous filament (or tape) is treated with resin and wound on a removable mandrel in a pattern.
fill	Yarn oriented at right angles to the warp in a woven fabric.
filler	A second material added to a material to alter its physical, mechanical, thermal, or electrical properties. Sometimes used specifically to mean particulate additives.
finish	A material with which filaments are treated containing a coupling agent to improve the bond between the filament surface and the resin matrix in a composite material. In addition, finishes often contain ingredients that provide lubricity to the filament surface, preventing abrasive damage during handling, and a binder that promotes strand integrity and facilitates packing of the filaments.
flame-sprayed tape	A form of metal matrix preply in which the fiber system is held in place on a foil sheet of matrix alloy by a metallic flame-spray deposit. Each flame-sprayed preply is usually combined in the layup stack with a metal cover foil or additional metal powder to insure complete encapsulation of the fibers. During consolidation, all the metallic constituents are coalesced into a homogeneous matrix.
flash	Excess material that forms at the parting line of a mold or die, or is extruded from a closed mold.
flexure	Any bending deformation of an elastic body in which the points originally lying on any straight line are displaced to form a plane curve.
fracture stress	The minimum stress that will cause fracture, also known as fracture strength.
fugitive binder	A resinous material used in the fabrication of metal matrix green tape (which see) preplies to hold the fiber system in place on the metallic foil sheet during shipping, storage, handling, and layup. During laminate consolidation, the fugitive binder decomposes and the products completely vaporize.

gel coat	A quick-setting resin used in molding processes to provide an improved surface for the composite; it is the first resin applied to the mold after the mold-release agent.
glass	A state of matter that is amorphous or disordered like a liquid in structure, hence capable of continuous composition variation and lacking a true melting point, but softening gradually with increasing temperature.
glass-ceramic	Solid material, partly crystalline and partly glassy, formed by the controlled crystallization of certain glasses.
green tape	A form of metal matrix preply in which the fiber system is held in place on a foil sheet of matrix alloy by a resinous fugitive binder (which see). Each green tape preply is usually combined in the layup stack with a metal cover foil to insure complete encapsulation of the fibers. During consolidation, the fugitive binder decomposes and the products completely vaporize.
hardness	Resistance of a material to indentation, scratching, abrasion, or cutting.
heat treatment	Heating and cooling of a material to obtain desired properties or conditions.
heterogeneous	Descriptive term for a material consisting of dissimilar constituents separately identifiable; a medium consisting of regions of unlike properties separated by internal boundaries. (Note that all nonhomogeneous materials are not necessarily heterogeneous.)
holography	A technique for recording and later reconstructing the amplitude and phase distributions of a wave disturbance.
homogeneous	Descriptive term for a material of uniform composition throughout; a medium that has no internal physical boundaries; a material whose properties are constant at every point—with respect to spatial coordinates (but not necessarily with respect to directional coordinates).
horizontal shear	The term "horizontal shear" is not approved nomenclature for the Guide, but it is included here for information only because of occasional encounters in the literature. Same as "interlaminar shear."
hot isostatic pressing	A form of ceramic, powder metallurgical, or ingot metallurgical forming or compaction process in which the mold is flexible and pressure is applied hydrostatically or pneumatically from all sides.
hot pressing	Forming a metal powder compact or a ceramic shape by applying pressure and heat simultaneously at temperatures high enough for sintering to occur.
hybrid	A composite laminate composed of laminae of two or more composite material systems.
impact strength	Ability of a material to resist shock loading.
inclusion	A physical and mechanical discontinuity occurring within a material or part, usually consisting of solid, encapsulated foreign material. Inclusions are often capable of transmitting some structural stresses and energy fields, but in a noticeably different (and less desirable) degree from the parent material.

injection molding	Forming metal, plastic, or ceramic shapes by injecting a measured quantity of the material into shaped molds.
integral composite structure	Composite structure in which several structural elements that would conventionally be assembled by bonding or mechanical fasteners after separate fabrication are instead laid up and cured as a single, complex, continuous structure—e.g., spars, ribs, and one stiffened cover of a wing box fabricated as a single integral part. The term is sometimes applied more loosely to any composite structure not assembled by mechanical fasteners.
interlaminar	Descriptive term pertaining to some object (e.g., voids), event (e.g., fracture), or potential field (e.g., shear stress) referenced as existing or occurring between two or more adjacent laminae.
interlaminar shear	Shearing force tending to produce a relative displacement between two laminae in a laminate along the plane of their interface.
internal stress, residual stress	A stress system within a solid (e.g., thermal stresses resulting from rapid cooling from a high temperature) that is not dependent on external forces.
interphase, interface	The boundary layer between the matrix and a fiber, whisker, or particle in a composite.
intralaminar	Descriptive term pertaining to some object (e.g., voids), event (e.g., fracture), or potential field (e.g., temperature gradient) existing entirely within a single lamina without reference to any adjacent laminae.
isotropic	Having uniform properties in all directions. The measured properties of an isotropic material are independent of the axis of testing.
lamina	A single ply or layer in a laminate made of a series of layers.
laminate	A product made by bonding together two or more layers or laminae of material or materials.
laminate orientation	The configuration of a crossplied composite laminate with regard to the angles of crossplying, the number of laminae at each angle, and the exact sequence of the lamina layup.
layup	A process for fabricating composite structures involving placement of sequential layers of matrix-impregnated fibers on a mold surface.
load	The weight that is supported by a structure, or mechanical force that is applied to a body.
mandrel	A form mixture or male mold used for the base in the production of a part by layup or filament winding.
matrix	The composite constituent that binds the reinforcement together and transmits loads between reinforcing fibers.
metal	An opaque material with good electrical and thermal conductivities, ductility, and reflectivity; properties are related to the structure in which the positively charged nuclei are bonded through a field of mobile electrons that surrounds them, forming a close-packed structure.

microstructure	The internal structure of a solid viewed on a distance scale on the order of micrometers. The microstructure is controlled by processing and determines the performance characteristics of the structure.
modulus of elasticity	A parameter characterizing the stiffness of a material, or its resistance to deformation under stress. For example, steel has a fairly high modulus, while Jello has a low modulus.
monolithic	Constructed from a single type of material.
near-net-shape	The original formation of a part to a shape that is as close to the desired final shape as possible, requiring as few finishing operations as possible.
nondestructive testing, evaluation	Any testing method that does not involve damaging or destroying the test sample; includes use of X-rays, ultrasonics, magnetic flux, etc.
orthotropic	Showing symmetry of properties in two orthogonal planes but different properties in the third plane.
phase	A region of a material that is physically distinct and is homogeneous in chemical composition.
plasticity	The property of a solid body whereby it undergoes a permanent change in shape or size when subjected to a stress exceeding a particular value, called the yield value.
polymer	Substance made of giant molecules formed by the union of simple molecules (monomers); for example, polymerization of ethylene forms a polyethylene chain.
pore, porosity	Flaw involving unfilled space inside a material that frequently limits the material strength.
prepreg	Fiber reinforcement form (usually tape, fabric, or broadgoods) that has been preimpregnated with a liquid thermosetting resin and cured to a viscous second stage. Thermoplastic prepregs are also available.
proof test	A predetermined test load, greater than the intended service load, to which a specimen is subjected before acceptance for use.
pultrusion	A fabrication process that uses guides, shape dies, and heat to produce long parts with constant cross sections, which are then automatically cut to desired lengths. The process consists of pulling dry fibers through a resin bath and then through steel dies that define the shape of the part and control the amount of resin in it. Heat must be supplied to cure the part.
radiography	The technique of producing a photographic image of an opaque specimen by transmitting a beam of X-rays or gamma rays through it onto an adjacent photographic film; the transmitted intensity reflects variations in thickness, density, and chemical composition of the specimen.
radome	A strong, thin shell made from a dielectric material, used to house a radar antenna.
refractory	Capable of enduring high-temperature conditions.
S-glass	A magnesia-alumina-silicate glass that provides very high tensile strength fiber reinforcement. Often regarded as the reinforcement fiber dividing "advanced" composites from reinforced plastics.

shearing stress	A stress in which the material on one side of a surface pushes on the material on the other side of the surface with a force parallel to the surface.
sintering	Method for the consolidation and densification of metal or ceramic powders by heating without melting.
slip casting, slip, slurry	A forming process in the manufacture of shaped refractories, cermets, and other materials in which slip is poured into porous plaster molds. Slip or slurry is a suspension of ceramic particles in water with a creamy consistency.
strain	Change in length of an object in response to an applied stress, divided by undistorted length.
stress	The force acting across a unit area in a solid material in resisting the separation, compacting, or sliding that tends to be induced by external forces.
structural materials or assembly	Those parts of a system that support most of the loading on the whole system.
substrate	Base surface on which a material adheres, for example a surface to be coated.
tack	Stickiness of a prepreg.
tensile strength, ultimate tensile strength	The maximum stress a material subjected to a stretching load can withstand without breaking.
thermal conductivity	The rate of heat flow under steady conditions through unit area per unit temperature in the direction perpendicular to the area—the ability of a material to conduct heat.
thermoplastic resin	A material containing discrete polymer molecules that will repeatedly soften when heated and harden when cooled; for example, polyethylenes, vinyls, nylons, and fluorocarbons.
thermosetting resin	A matrix material initially having low viscosity that hardens because of the formation of chemical bonds between polymer chains. Once cured, the material cannot be melted or remolded without destroying its original characteristics; examples are epoxies, phenolics, and polyimides.
toughness	A parameter measuring the amount of energy required to fracture a material in the presence of flaws.
tribology	The study of the phenomena and mechanisms of friction, lubrication, and wear of surfaces in relative motion.
ultrasonic testing	A nondestructive test method that employs high-frequency mechanical vibration energy to detect and locate structural discontinuities or differences and to measure thickness of a variety of materials.
unibody	Integrated structure containing the chassis as well as elements of the body of an automobile.
unitized design	A unitized design refers to the philosophy of designing and building a structure in one piece, thereby eliminating the need for fasteners. Traditionally these structures were manufactured in separate pieces

	and then mechanically aligned, shimmed, and fastened. The unitization of the piece then can save much of the alignment and assembly work and makes the piece less likely to contain undetected flaws associated with drilling holes, etc.
viscoelasticity	Property of a material that is viscous but also exhibits certain elastic properties such as the ability to store energy of deformation, and in which the application of a stress gives rise to a strain that approaches its equilibrium value slowly.
wear	Deterioration of a surface due to material removal caused by relative motion between it and another material.
wettability	The ability of any solid surface to be wetted when in contact with a liquid.
yield strength	The lowest stress at which a material undergoes plastic deformation. Below this stress, the material is elastic.

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